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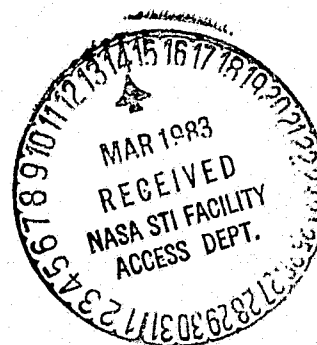
# RESEARCH PRIORITIES FOR ADVANCED FIBROUS COMPOSITES

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16. Abstract  Priorities for research in advanced laminated fibrous composite materials are presented. Supporting evidence is presented in two bodies, including a general literature survey and a survey of aerospace composite hardware and service experience. Both surveys were undertaken during 1977-1979. Specific results and conclusions indicate that a significant portion of contemporary published research diverges from recommended priorities.  <div style="text-align: center;">ORIGINAL PAGE IS OF POOR QUALITY</div>					
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FOREWORD

This project by nature has enabled us to delve into an enormous pool of aerospace resources. We are especially grateful to our NASA LeRC sponsors for supporting us with NSG 3172 and NSG 3172 Supplement A.

Together, we wish to thank all of our friends in the aerospace community for willingly sharing with us their knowledge, experience and information, and for giving us their valuable time and help. It has been a luxury to have been generally received with such helpful and willingly open attitudes. In light of the potentially sensitive and proprietary nature of some of the information collected, it is most appropriate to omit listing of individuals and their affiliations. We regret this omission of personal acknowledgments, and trust that it will be understood in such circumstances. We are grateful for permission granted to use figures and photos. Many are adapted or reprinted by permission of the American Society for Testing and Materials, Copyright.

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\*Much of the learned service experience information is sensitive and is therefore reported only in a generic fashion in "Service Experience."

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## SUMMARY

This document recommends research priorities in advanced fibrous composites for the aerospace industry. The supporting material is presented as state-of-the-art reviews of:

1. Service experience
2. Technical literature.

Aerospace structural service experience has been collected through personal visits, numerous telephone calls, a written survey, attending conferences, a short course, and committee meetings over the period lasting from September 1977 to June 1979. Aircraft for which information has been gathered are listed in Appendix A. The literature through 31 December 1978 is reviewed from a material property standpoint. This literature is listed in the Bibliography.

To resolve a number of ambiguities found in vocabulary usage in the literature, we have found it useful to create a working vocabulary for the purpose of writing the report. Appendix B presents this vocabulary.

In collecting material for this report, we have determined

- design and reliability assurance methods;
- test procedures and their resulting data;
- problems and failure modes encountered in service; and
- the character of the research literature.

It is our conclusion that new research priorities and objectives would be advisable if this work is to serve the aerospace industry's use of composites. The situation is particularly acute in the area of material characterization and testing. In this connection, we are reminded of a nearly twenty-year-old statement made by the ASTM Committee on Fracture Testing of High-Strength Sheet Materials:

In principle, there are two ways to prove a test method. With a large body of service performance data available - particularly service failure data - a correlation can be attempted between test result and service performance to determine if the test method is measuring the material characteristic that governs performance. In the absence of adequate service performance data, it may still be possible to establish a test method if a suitable method of analysis is available by which it may be shown that the test method is measuring the significant quantities governing performance and that the test result may be generalized to the more complex conditions existing in an actual structure.

Our studies show that a weak correlation appears to exist between much of the experimental test data and service performance; our recommendations are aimed at resolving such difficulties.

## INTRODUCTION

This report contains constructive criticism aimed at advancing the state-of-the-art of composite material structural integrity assurance.

It seems that a great deal of research is devoted to these relatively young materials, yet only limited use is being made of composites in primary aerospace structure. We feel that this calls for an examination of the nature of the ongoing research.

Our examinations of the literature and service experience up to December 1978 are presented as state-of-the-art reviews. By comparing the two, we have developed and recommended specific priorities for future research.

## STATE-OF-THE-ART REVIEW

### Introduction

Our state-of-the-art review of advanced fibrous composite materials is divided into two tasks: (1) collection of service experience information, and (2) the literature review.

The review is a collection of written and verbal information obtained through libraries, telephone conversations, personal visits, conferences, committee meetings, a short course, and a written questionnaire. We feel that the domestic aerospace community has been sufficiently canvassed in obtaining state-of-the-art information. However, in view of the size of the aerospace community and the accumulated body of knowledge, it must be recognized that not every bit of pertinent information could have been collected. It should be noted that some of the information we have been able to acquire is either proprietary or potentially sensitive. Where appropriate, we respect the privacy of our sources.

We have in addition examined some literature and service experience concerning earlier, less advanced composites. This has allowed us:

1. To gain historical perspective
2. To recognize that earlier materials are still being used in flying structure
3. To aid reporting on hybrids which in some cases mix advanced with "unadvanced" materials
4. To force more of the report to deal with materials which might serve as future alternatives in case the carbon fiber hazard

study resulted in the eventual demise or severely limited future use of carbon (graphite) fibers.

For Task #1, the collection of service experience information, we choose to focus upon activities in the domestic military and commercial aircraft community. This is done in order to trim the task to a manageable size with our given resources. Although there is increased use of composites within the automotive, sporting goods, agriculture, housing, other defense, energy, electronics, marine, and other industries, we feel it is the aerospace community which is generally at the forefront of the technology.

A perusal of Aerospace Applications, Appendix A, will show that military use of composites tends to push the state of the art more than commercial aircraft useage. In particular, note that there are a number of military craft having both primary and secondary structural applications of composites, whereas most commercial useage has been secondary structure. This appears to result from several reasons:

1. Military craft, particularly fighters, are generally higher performance machines needing higher technology than commercial aircraft. They are subject to more severe loads, higher flight speeds involving aerodynamic heating, and may receive less rigorous maintainance.
2. Commercial craft safety and the associated liabilities are strongest concerns. New technologies are introduced very conservatively in this atmosphere.
3. Commercial manufacture is more profit oriented. Cost competitive parts are produceable only after the composites learning curve is overcome. Unfortunately this curve is often associated with high changeover costs and time.

In addition to domestic military and commercial aircraft, some spacecraft, missiles, private aircraft, and non-domestic service applications have been discovered and are included. In some applications it will be noted that very limited information has been dug up, nonetheless such applications are listed for their pertinence. In many applications, much service information is sensitive and has to be omitted.

Several excellent reviews pertinent to both tasks precede ours. These previous reviews of service applications (Hackman 1973, Hardrath 1977, Hodges 1977, Pride 1978, Salkind 1973 and 1976) are incorporated into this report. To these we have added by generally being nosey and doing a lot of sniffing around, hopefully in a friendly and sincerely constructive context.

The written survey has been the least successful method in getting at the desired service experience information. In many instances participants felt it was superficial after having talked with us. We in fact found conversation to be most rewarding. Also, recipients may have been hesitant to put their answers in writing. The survey was, in any event, a very useful means of organizing some of our thoughts, developing categories for damage assessment, and developing questions and questioning methods to use in conversation.

The review is therefore in two parts. Part I is the service experience review which will immediately follow. Generic issues are reported based upon information collected. This review relies upon particular aerospace service applications listed alphabetically in Appendix A. Part II is the literature review. It is organized in the framework of the effects of various excitations upon damage assessment parameters.

## Service Experience

Papers concerning design and service experience are referenced in Appendix C under the appropriate application.

## Summary of Advantages

The accumulated service experience and the desire for increased use of composites in aircraft structure demonstrate the following specific advantages over traditional materials:

Light Weight - Advanced composite structure is typically 20-30% lighter than the equivalent metal alloy structure it replaces. Furthermore, when a large percentage of flight vehicle weight is designed using composite materials, there is an extra weight savings generated by the structure downsizing. The downsizing effect follows decreased requirements for wing areas, structural strength, lessened powerplant requirements, etc. This effect has not yet been fully realized in commercial or military aircraft, however it has been demonstrated as a real effect in many applications. Consequently, as greater percentages of an aircraft's structure are initially designed using composites, rather than composite parts being tagged on as replacement parts for metal, this advantage will become more pronounced.

Weight savings is a fundamental advantage leading to aircraft fuel savings. In recognition of this, NASA is actively pursuing increased use of composites as part of the Aircraft Energy Efficiency (ACEE) program. A number of service applications cited in this report [DC-10 upper aft rudder, L-1011 ailerons, B-727 elevators, L-1011 vertical fins, DC-10 vertical fin, B-737 horizontal stabilizer, engines] are associated with the ACEE program. An excellent overview of the program is offered by Hardrath, 1977.



### Cost

Reduced costs using composites rather than metals is possible and is demonstrated in many applications. This advantage follows the inherent nature of composite fabrication as well as service improvements.

Use of traditional composites typically demonstrates cost savings, whereas use of advanced composites has sometimes indicated increased expense over traditional metals. Despite such early indications, advanced composites have proven capabilities of being less costly than traditional metals. The early indications of high cost are attributable to shop changeover costs, initially high fiber prices which are decreasing, and composites learning curve climbing in general.

Cost savings follows the great reduction in parts count brought about by capabilities of molding large subassemblies. This results in less fasteners, lower tooling costs, lower labor and fabrication costs, lower handling costs, less machining, and reduced scrap (5% compared to 80% Hackman 1973) in structure which must often have compound curvature.

Once in service, added cost savings often follow from enhanced service capabilities such as reduced corrosion.

### Corrosion Resistance

Traditional composite materials have shown corrosion resistance superior to aerospace metal alloys in many applications. This advantage has been demonstrated by glass and Kevlar\* composites, which exhibit little if any electrochemical corrosion. Some galvanic corrosion of carbon fiber composites has been experienced, in the presence of aluminum; however no problems with boron composites were discovered.

\*Dupont registered trademark.

Degradation from ultraviolet sunlight radiation and from moisture penetration can occur, but these effects can be more readily controlled and protected against than can metal corrosion.

### Stiffness

High, tailorable stiffness combined with light weight is a proven advantage of composites. The advanced composites using fibers such as Gr and B have enabled structural designs to be stiffer, smaller, and thinner than comparable metal structure. Increased aspect ratios are possible. This has been a great advantage in structure which often must be stiffness-designed against flutter instabilities.

The increased specific stiffness of composites has enabled lift and control surface cross sections to be thinner and smaller, thus enabling more effective structure which offers less drag at fighter air speeds.

Aeroelastic tailoring has been enhanced by composites. The designer is able to design the material to warp or twist under bending or extensional load for improved performance [HiMat] on a par unobtainable using metals.

### Dimensional Stability

The near-zero thermal expansion coefficient achievable in carbon fiber composites, combined with the other advantages, makes them attractive in applications such as spacecraft reflectors, where thermal distortion must be minimized. One side of spacecraft structure may experience high temperatures due to impinging sunlight, whereas the dark side may be near absolute zero.

### Improved Reliability and Serviceability

Composites have proven to be more reliable and serviceable than metals

in helicopter rotor blades. This follows from lessened notch sensitivity, improved corrosion resistance, and improved fatigue life over comparable metal blades. Many secondary applications have demonstrated lessened maintenance requirements, due to reduced corrosion resistance.

### Damping

Composite materials have inherently high vibration and noise damping properties. This can reduce noise and vibration amplitudes (Hardrath and Dexter 1977).

### Reduced Drag

Elimination of fasteners from large molded structure enables smoother surfaces to be built, decreasing drag (Hardrath and Dexter 1977).

### Deterioration in Service

Deterioration of composite structures in service has occurred; except in a few isolated and rare incidents, failure has not occurred. In general, deterioration has either not occurred or if it has, it is of only minor consequence. In most applications, composite structures have been damaged or deteriorated at a slower rate and generally outperformed comparable metal structure. Composite structures collectively have demonstrated capability to perform in most applications without failure or critical damage. This history reflects the conservative philosophies used in introducing composites to aerospace structure, and also the potential capabilities and strengths of these materials.

Specific modes of deterioration are evident in the service experience. Damage, as we have defined it, is often neither reported nor measured; however, deterioration of one sort or another is typically described. From such deterioration one could presumably infer damage accrual, if the proper

correlations were available. Usually in service, the deterioration is repaired locally or the structure is replaced, often without knowledge concerning the deterioration's effect upon subsequent performance of the part.

Service experience gives evidence of the following modes of deterioration:

- Delamination and blistering
  - Erosion
  - Corrosion
  - Cracking (other than delamination)
  - Debonding of composite skins from core material
  - Debonding of bonded joints from metal
  - Fastener hole enlargement
  - Gouging (removal of material)
  - Denting, in which the core may be crushed and the skin develops waviness
- Specific causes of deterioration are evident in the service experience.

• Ground Handling by service personnel has proven to be a major cause of deterioration in service. We haven't studied metal structure ground handling deterioration, and can't make any conclusive comparison of susceptibility. The modes of deterioration brought on by ground handling vary quite a bit. Dropped tools have caused delaminations, cracks, dents, debonds. Service personnel walking and kneeling upon aircraft during the course of regular maintenance have in some cases created debonds of skins from honeycomb core.

Improper or incomplete installation of secondary structure has resulted in actuator fittings being pushed through the structure, rather than actuating it. Dropping of composite structures has caused delaminations and cracks, particularly if the corner of a flat part impacts a concrete runway.

Tools and fasteners accidentally left inside cavities into which composite structures withdraw have cracked and penetrated thin skins. Cables drawn across composite edges have chewed into the edge.

Forklift trucks have impacted thick skins and leading and trailing edges, created gouges, dents, and penetration holes.

A number of incidents have resulted from errors which we judge would have also damaged or caused failure in comparable metal structure. We have not reported these specific incidents for reasons given in State-of-the-Art Introduction.

We have learned privately that at least one such incident may have been the reason for failure of a secondary structure during flight. People who push for the enhanced and increased use of composites claim that Murphy's Law has been at work, and that as a consequence composites have developed a bad and undeserved reputation amongst pilots. We have not undertaken to poll pilots for their opinions on composite structure. We would in any event be prepared to argue for increased use of composites based upon their many advantages (see Service Experience) and the large body of accumulated successful service experience.

• Impact Loads (FOD\*) due to birds, hail, stones, runway debris, and dropped tools have caused delamination, and cracking. There is mixed experience concerning whether or not FOD typically deteriorates thick sections. Heavy impact loads, more properly collisions, apparently due to pilot errors have caused extensive damage. In turbine blades, hard body impact of sufficient magnitude usually results in very localized cracks and debonds, whereas soft body impact results in larger areas of deterioration, and, even in deterioration at locations removed from the area of impact.

\*Foreign Object Damage

- Hygrothermal Excitations, sometimes in combination with other excitations and design and manufacturing variables, have demonstrated significant effects upon many structures. Moisture entering aluminum honeycomb cores can promote corrosion. Moisture has promoted debonding of composite to metal structure. It is difficult to pin down specific structural deterioration to moisture and temperature alone. The combined excitation of moisture and temperature lowers epoxy  $T_g$  to an extent greater than each excitation alone. Operation of structure below epoxy  $T_g$  can deteriorate stiffness undesireably. For this reason, it is commonly attempted to operate above  $T_g$ , which is typically near 250°F or 350°F before hygrothermal excitations.

- Skydrol Hydraulic Fluid in combination with other in-service excitations caused delaminations of polysulfane matrix composites [B737 spoilers].

- Service Loads. We have learned privately that there may have been one incident where insufficient stiffness led to flutter instability of a composite part.

- Design and Manufacturing Variables and Flaws built into parts such as voids, excessive resin areas, starved areas, debonds, (Hoffman and Konishi 1977), have occurred, and have sometimes caused deterioration in service. Parts have been crushed due to thermal mismatch with dies.

### Problems

A number of generic difficulties have arisen in developing applications of composites. Many of these have been solved or successfully dealt with, while others remain problems at least in some portion of the community. In coming up with recommendations for future research, the service applications and their associated manufacturers have presented the following problems and difficulties:

- Prepreg Inconsistency upon receipt from suppliers and due to aging in storage is experienced by many aerospace manufacturers. Furthermore, the problem persists due to lack of prepreg user prepreg Q/A\* and to the proprietary nature of the prepreg business.

- High Costs have resulted from a combination of initially high material costs, shop changeover costs, and production problems. In some cases certification procedures dictate certain processes during production. The cost problem is essentially one of climbing the composites learning curve, and does not persist. In fact, much recent emphasis on cost savings in industry has resulted in cost-savings over metal structure. The problem of certification requirements being intermixed with manufacturing persists, however. Epoxy cure cycles are part of certification so companies are very reluctant to change. Polymer chemists have programs to develop epoxy specifications and pin down variabilities.

Production problems which result in damaged parts, excessive voids, and resin rich areas have been solved by some manufacturers and not by others. Thermal expansion coefficient mismatch with dies can be troublesome in producing large parts. As a large Gr/E part cools down inside a metal mold initially at 350°F, the mold can shrink enough to cause compressive failure in the part. The thermal mismatch problem is also problematic when attempting to bond composite to metal structure at high temperature.

- Joining Methods problems arise due to the extent to which holes penalize strain allowables [Bl longeron] and due to difficulties with bond reliability..

\*Quality Assurance.

- System Integration has proven difficult in a number of applications (B1 vertical stabilizer, F14 wing, etc.). Problems meeting antenna, shielding, lightning protection, electrical grounding, and deicing requirements arise due to the electrical nature of finished form composite laminates. Although Gr fibers are highly conductive, when embedded in an epoxy matrix they become well insulated by the matrix. This has sometimes required special developments for lightning protection such as outer layer conductive foils or screens. Also, the laminate may then need special fasteners if it is to serve as a grounding structure for electronic equipment.

- Design Problems are an impedance to increased primary structural application of composites. Experienced specialists in composites are often required. There is a lack of design methods and failure prediction criteria, adequate data bases, and accept/reject criteria for manufactured and in-service parts, subject to moisture and temperature effects. The design task now has the added dimension of requiring design of not only the structure but of the material itself. There is a basically new thing to be considered in the design of composite aircraft structure that is generally neglected in metallic structural design. That is, the application of major loads such as bending on the wings produces transverse tensile and interlaminar shear loads, what might be called secondary loads. In metallic structural design these loads are generally neglected and seldom calculated because the homogeneous and isotropic material, and the fasteners, can generally carry these loads. However, in composite structure, the nonhomogeneity and anisotropy can cause problems when these loads are applied. They often act normal to principal strength directions and may in fact act in the laminate's weaker or most vulnerable direction. So they must be considered and designed for.



The effects of delamination, voids, porosity, and other defects on the residual strength of a structure is a primary concern (Porter and June 1978). Sources of delamination include manufacturing imperfections, impact, and in-service loading generated delaminations. More knowledge is needed about growth characteristics of flaws, particularly delaminations under service loadings. There is presently no standard criteria on designing large cutouts. The effects of real load spectrums on structural life are unknown, and the results of accelerated test techniques are questionable until a real-time data base is accumulated.

Designers wishing to build fuselages find that post-buckling data is almost non-existent. Means of assessing crashworthiness and repairability are lacking.

• NDE Methods leave much to be desired. The inherent anisotropy and nonhomogeneity cause much difficulty when using acoustic techniques. NDE during all stages of material receipt, manufacture, and service have been expensive and give questionable indications. There is disagreement concerning whether or not damage in service from FOD is usually visible in thinner skins. Besides the difficulty experienced in finding and identifying flaws and damage, much difficulty is encountered when confronting the issue of accepting, rejecting, or repairing the structure.

• Repair Methodologies have been successfully developed mostly for minor problems in secondary structure but not for primary structures. The transfer of load into a patch on a thick laminate presents much difficulty in that fastener joints penalize strain allowables while bonded joints are of questionable reliability.

• Failure Analysis is just beginning in composites. Given a part that has failed, one would like to be able to say why and where failure initiated, in a manner similar to metals technology, so that improvements could be made.

Some work on determining crack propagation directions has been done based upon hackle directions. Discussion at a recent conference centered upon the issue of whether failure initiates at one site or at multiple sites. The broomed appearance of many failed specimens indicates that many surfaces may be generated during failure of composites.

- Unanticipated Damage Accrual occurred in some early composites applications, such as that due to moisture and temperature excitations. These for the most part are now anticipated but not always easily quantified, especially in the context of being combined with other excitations, flaws, and material design and manufacturing variables.

## State-of-the-Art Structural Design and Reliability Assurance

A sense of how state-of-the-art structural design, manufacture, and reliability assurance proceeds is given. In general, the composite design and manufacturing process is dramatically different from metals. Immediately, two points are of interest. First, in composites, there is need for both structural design and material design. Secondly, a major disadvantage of the current process is that the bottom line to assure reliability is often an expensive, time consuming full scale test of structure or subcomponents and periodic in-service inspections. This need arises from the inability to assess by predictive techniques whether or not all structural design criteria have been met.

Early concerns associated with designing replacement composite structure to be lighter than metals have given way to concerns about reduced costs, reliability, safety, and maintainability of original equipment, not just replacement, structure. We will now march through most of the process and discuss the tools and data base currently available and in use. We shall exclude discussion of variations and details of the process, particularly manufacturing processes, many of which are proprietary. These details are important but not central to this discussion. Instead, the aim is to outline broadly the current procedures as a basis for pointing out deficiencies and problems.

We begin by assuming that the need for a structure is indicated. The particular structural design factors\* must be then identified. In aircraft structure, reliability, stiffness, strength, and weight are often dominant design factors. Relative degree of importance of each factor depends largely upon whether the structure is intended for military or commercial usage. Once each structural design factor is quantified or constrained it creates a structural design criteria. For example, stiffness must have sufficient

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\*See Appendix B.

margin to prevent flutter instability. Cost must be constrained for competitive purposes. Unfortunately, reliability criteria are not so simply stated. Such criteria are set to some extent externally and are generally in the infancy stage of development.

At this point composite engineers with extraordinary expertise (often called the wizards) do their artwork. There are perhaps one hundred such composite engineers in the community who are able individually or in small teams to take good, largely confident, first shots at designing composite structure. Drawing upon their collective experiences, data, and know-how, and working within their community's capabilities and constraints, such designers are able to envision in reasonable detail a combination of structural configuration, dimensions, geometry, materials, and manufacturing process, that will be pretty close to a workable, efficient, and reliable structure. More detailed structural and laminate design, analysis, and optimization tasks may in many cases be assigned to an increased number of individuals, often under the watchful eyes of the wizards.

The structural design process using composites has far greater complexity than metal alloys, because of the additional tasks associated with composite material design. In addition, there is increased interdependence between design, manufacturing processes and material choices. The wizards are efficient at cutting through these complexities, at knowing what to assume and what not to assume, and what tools to use along the way.

What procedures and techniques, and tools and data are being used for design purposes? They are found to vary somewhat from the commercial to the military context, from company to company, from structure to structure, and so on. Nonetheless certain similarities exist and may be discussed.

The procedures and techniques used for structural and laminate design fall into three categories:

Analytical procedures use rigorous, analysis-based theoretical models to predict behavior of the whole from constituent interactive behavior. Thus the whole and its constituents may either be the lamina and its fibers and matrix, or the laminate and its lamina. In essence, analytical procedures may rely upon micro, macro, or both micro and macromechanical\* analysis. The predictions may or may not be confirmed by laboratory tests.

Semi-empirical procedures blend theory and experiment using models which have some theoretical physical basis but rely upon disposable constants to correlate the model with a data base of measured lamina or laminate properties. Again, the models may be micro or macromechanical.

Empirical procedures rely upon direct testing of structure to determine performance, and have no theoretical base. Curve fitting and correlation are commonly attempted to organize the data and hopefully to be used in developing empirical models.

In practice, a mixture of the three design procedures is used. The mixture depends upon the particular structure being designed, the design team, the available data base, and the accuracy and limitations of available models. Often it is the case that all available predictive procedures and tools fall short of modeling all the design conditions, even common and relatively simple design conditions. This often requires simulated structural testing (empiricism) as the bottom line. Already, some predictive and correlative procedures are in use. It is interesting at this point to note that, in the literature, many attempts are made at accurately predicting laminate and laminate behavior using purely analytical procedures. Such micromechanical analytical techniques have been found to be deficient in accurate prediction of most lamina properties which are required for design purposes. However,

\*See Appendix B.

micromechanical analytical techniques are used to provide a qualitative rationale for designing and developing materials. The most useful micromechanical analytical models are based upon simplifying assumptions which facilitate mathematical analysis and comparison of proposed designs. The numerous assumptions required therefore should provide meaningful models which are not necessarily highly accurate and correct models.

For material design purposes, the semi-empirical micromechanical Halpin-Tsai equations are quite useful in studying lamina properties  $E_1$ ,  $V_{12}$ ,  $E_2$ , or  $G_{12}$  (Jones 1975) as affected by changes in constituent properties and fiber volume fraction.

Both semiempirical and analytical macromechanical models are in more widespread design use. On the lamina level, generally accepted lamina stress-strain analysis relations are in widespread use, and rely upon invariant properties of orthotropic lamina. However, there are several strength models in use, including maximum point stress theory, maximum strain theory, and Tsai-Hill (also called Hill, von Mises, and Distortional Energy). A paper is available (Sendekyj 1972) which reviews various strength theories.

On the laminate level, classical lamination theory or analysis is in widespread use for designing based upon stiffness and strength. Invariant laminate properties are also used.

Analytical design techniques in use therefore rely upon both lamina and laminate macromechanical models. The models are often computerized and typically use finite element methods in an iterative fashion (Jones, pp. 190-232). The iterations may be made using static strength, stiffness, and weight criteria. Computerized methods have been used to select optimum material designs, and efforts have addressed streamlining and simplifying the composite structure design process (Hackman, pp. 55-61). Fastened joint design, small hole design, and small cutout design for simple situations currently proceed using fracture mechanics technology and/or maximum strain allowables when data is available.

Empirical procedures without models are currently used for many design purposes due to lack of models and data. Fatigue strength of structure with hygrothermal excitations, material design and manufacturing variables, flaws, and in-service real spectrum loads proceed empirically. Often, empirical techniques are used to supplement other design techniques in situations involving complex and even simple combinations of flaws, excitations, and material design and manufacturing parameters. Empirical techniques are seldom used exclusively to design structure, however they are often used to check the wizard's design and to assure reliability. Such empirical techniques are typically extremely expensive and time consuming. As such, there exists great desire to develop techniques which reduce the need for empirical tests specific to one's particular structural application.

Currently very few empirical or semiempirical models have been developed. The most effective procedures will enable structural and laminate design to proceed by extrapolation, interpolation, and/or prediction using a common philosophy and data base.

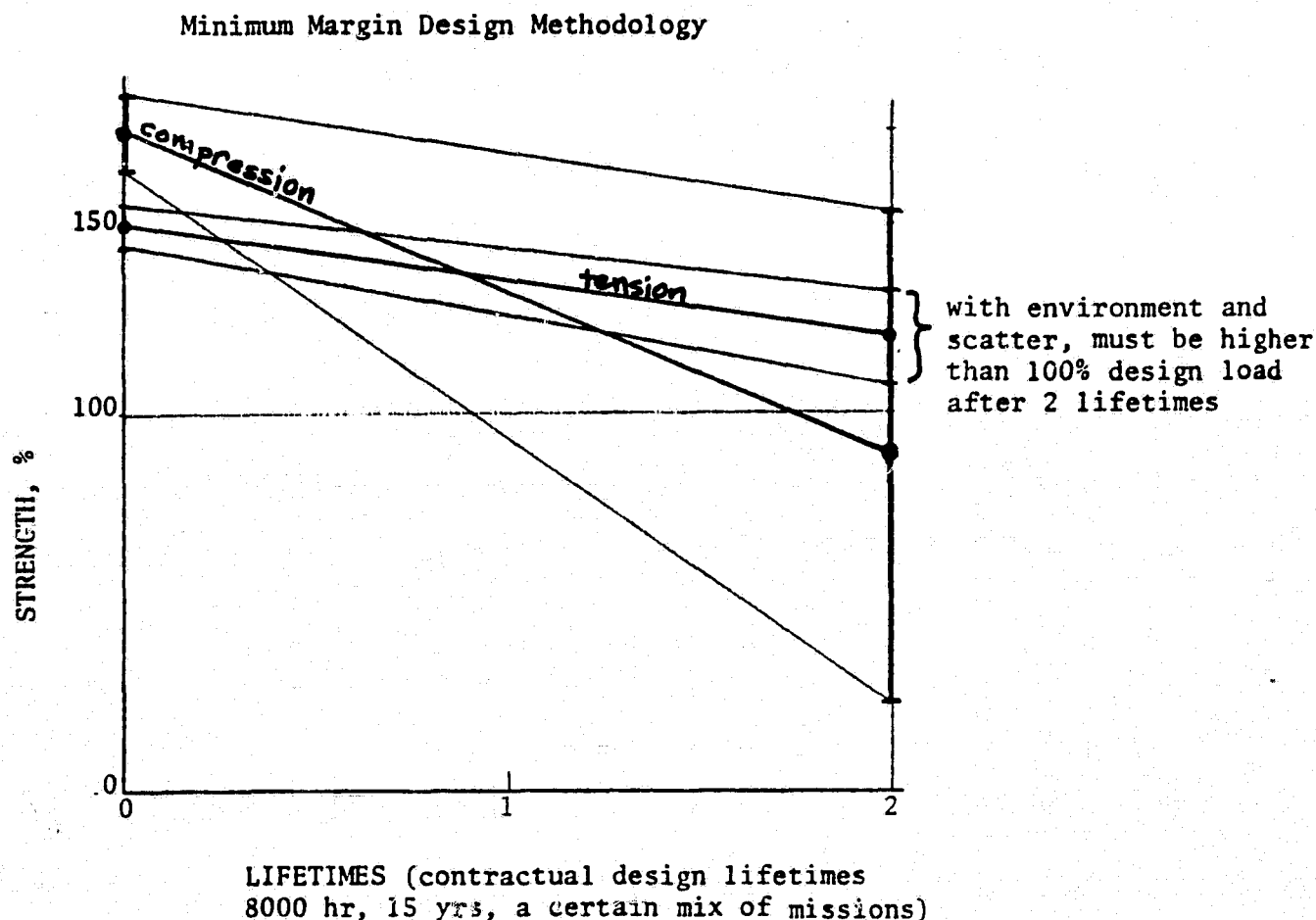
At this point, after design, the structural design criteria, and particularly reliability, are not yet fully ascertained. How then is reliability assured? Reliability is often assured by a combination of design, Q/A\*, full-scale structure or component testing, periodic in-service inspections, and in some cases repair or replacement. Current Q/A methods often rely upon arbitrarily set disposition criteria on flaws resulting from the entire material manufacturing process.

It is noteworthy that semiempirical macromechanical techniques relying upon a measured data base used during the design process are often powerful tools when applied to the task of assuring integrity and reliability, but are much cheaper and more effective than purely empirical methods. However, current semiempirical

\*Quality Assurance.

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techniques are rather limited in scope of excitations, flaws, material design and manufacturing variables, and material parameters which may be treated. It is easily envisioned that semiempirical macromechanical models are more easily extended to accurately model real situations than are micro-mechanical or purely theoretical macromechanical models. Q/A techniques currently involve attempts at assuring that prepregs have consistent properties, developing monitoring of and control of manufacturing processes, and inspection of the finished structure. Inspection often involves NDE, specifically ultrasonic c-scan. However, critical flaw levels are often arbitrarily set. Proof load methodology, or minimum margin design, has been tried for Q/A. It has been found to have a serious deficiency in certain applications. This is best discussed using the following figure.





The problem is that failure modes other than the one that is initially proof tested may take over and dominate during the lifetime. In the above example, proof testing at 150% of tensile ultimate load is made. This would guarantee acceptable tensile strength over the lifetime taking into account environment, scatter, and tensile strength damage accrual. However, the problem is that compressive strength damage accrual may cause structural failure because it accrues at a faster rate than does tensile strength damage. This may occur despite the fact that compressive strength was initially much higher. Considering a structure such as a wing having upper and lower skins, each having respective fatigue loadings generally in compression and tension, one would then initially desire to proof test the upper surface in compression. However, the desired magnitude of the upper skin compressive load, which is feasibly produced only by loading the wing in bending, is such that the proof load will cause tensile failure in the lower skin long before the desired compressive magnitude is reached. Axial loading of the wing to produce only compression on both skins isn't viewed as being feasible.

NDE methods in service have been studied and documented by M. L. Phelps 1979. In summary, ultrasonic c-scan, coin-tap, visual inspection and other techniques are being used. Problems including expense and lack of thoroughness exist, and requirements for effective NDE development are given.

### NDE In Service

An excellent state-of-the-art review of NDE methodologies used in service for commercial composite structures is available (Phelps 1979). We reproduce some of Phelps' report here. It is noteworthy that Phelps' report also contains information concerning other NDE methods, not all of which have found their way into commercial service. In general one finds some discussion of NDE in the majority of all the experimental literature. The recent 10,11 October 1978 ASTM Symposium on Nondestructive Evaluation and Flaw Criticality for Composite Materials is indicative of the volume and intensity of activity in this area.

State-of-the-art NDE for military composite hardware may generally use some more advanced techniques (personal communication with Phelps), since more production composite structure is in service on military aircraft.

We present here (pp. 36-60) the summary page of Phelps' report and his Appendix B. We encourage NDE researchers also to consider the recommendations and highly pertinent information presented elsewhere in Phelps' report.

## SUMMARY

A survey was conducted to determine current in-service inspection practices for all types of aircraft structure and specifically for advanced composite structures. The survey consisted of written questionnaires to commercial airlines, visits to airlines, aircraft manufacturers, and government agencies, and a literature search.

Existing inspection methods and equipment for in-service inspection of aircraft structures are documented in this report. A reference in-service inspection baseline and preliminary in-service inspection program for advanced composite structures on commercial transport aircraft have been documented and are appendices to this report.

With the data obtained in Phase I, a Phase II plan has been prepared for development and improvement of in-service inspection methods for graphite-epoxy composite aircraft structures and presented to NASA-LRC for approval.

(Phelps 1979)

**PRELIMINARY  
INSPECTION PROGRAM FOR  
ADVANCED COMPOSITE STRUCTURES ON  
IN-SERVICE AIRCRAFT**

**APPENDIX B  
TO  
PHASE I REPORT**

**"ASSESSMENT OF STATE-OF-THE-ART OF  
IN-SERVICE INSPECTION METHODS FOR  
GRAPHITE EPOXY COMPOSITE STRUCTURES  
ON COMMERCIAL TRANSPORT AIRCRAFT."**

**CONTRACT NAS1-15304**

**Prepared for**

**NATIONAL AERONAUTICS AND SPACE ADMINISTRATION  
LANGLEY RESEARCH CENTER  
Hampton, Virginia**

Appendix B  
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Appendix B  
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## 1.0 INTRODUCTION

This document was prepared under NASA Contract NAS1-15034 and fulfills task (e) the contract Phase I documentation requirements. It establishes the inspection methods, particularly nondestructive testing (NDT), that have been or currently are being used to inspect advanced composite structures on in-service aircraft. The data for this document were obtained from a literature review of articles, written questionnaires to commercial airlines, and visits to 14 airlines, three manufacturers, and three government agencies.

## 2.0 SCOPE

Each inspection method that has been used or is in current use on advanced composite structures on in-service airplanes is included in this document. Inspection methods that have not been used in service are excluded. For example, a particular method may have been evaluated and determined to be acceptable in a laboratory evaluation of a part removed from an in-service airplane. However, this method would be excluded, as confidence in it has not been demonstrated by using it for flight-status aircraft.

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### 3.0 INSPECTION METHOD SELECTION

#### 3.1 PRE-INSPECTION DATA

Determine:

1. Structural areas or items to be inspected, their location, geometry, and material identification.
2. Details of the underlying structure, including modifications.
3. Type and probable location of potential defects. Acceptable defects detected in fabricated parts prior to installation on the airplane should be identified for type and location.
4. Removals required for access to inspection area.
5. Safety precautions or special requirements.

#### 3.2 METHOD SELECTION

1. Refer to table 1 for selection of appropriate inspection method. Section 4.0, Defects, and section 5.0, Inspection Methods should be consulted for details.
2. A "Primary Method" normally would be used for initial inspection and the "Secondary Method" would be used if special circumstances prevent use of the Primary Method.
3. Use either category method for backup verification of defect and evaluation to determine size, type, and location.
4. The category "Potential Application" indicates that the method may detect the type of defect in question, but not enough evidence of capability or details of the specific technique are available.
5. "Not Applicable" is indicated when this is obvious by the nature of the method or if recent experience on advanced composite structures have demonstrated lack of capability.

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#### 4.0 DEFECTS

In-service defects as listed in table 1 are defined as follows:

1. Blisters--localized raised areas--may result from a delamination or disbond.
2. Scratches--visual surface scratches--readily detected on painted surfaces.
3. Dents--localized depressions--may indicate damage beneath the surface.
4. Delaminations--separation between plies in a laminated epoxy/fiber composite part or at the bondline of two separately identified parts.
5. Disbonds--separation at the bondline of parts joined by an adhesive bond.
6. Cracks--material separation in the resin matrix or across plies rather than between plies.
7. Impact Damage--shattering of matrix, small cracks, broken fibers, etc. resulting from impact of an object on part surface.
8. Fastener Hole Damage--cracks, small delaminations, matrix shattering, hole elongation, etc. caused by shear loads at the fastener-to-hole surface interface.
9. Fastener Pull-Through--fastener head has pulled into or beneath the surface of the part.
10. Lightning Damage--burns, matrix shattering, fiber damage, etc. resulting from lightning strike.
11. Honeycomb Core Damage--separation of honeycomb walls, crushed core, corrosion, etc. resulting from impact, entrapped water, and other causes.
12. Water-in-Honeycomb--entrapped water that may cause damage through freezing, thawing, corrosion of aluminum honeycomb, etc.



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13. Moisture Absorption/Degradation--absorption of water by graphite-epoxy parts in high humidity environments has resulted in degraded mechanical properties.
14. Heat Damage--overheating due to localized fire or overheated hardware may cause discolored or scorched paint and charred surfaces.
15. General Surface Deterioration--any gradual deterioration that may result from severe environmental or service conditions.

## 5.0 METHODS

### 5.1 VISUAL AND VISUAL/OPTICAL

#### 5.1.1 EQUIPMENT

Lights, low-power magnifying glasses, rigid and fiber-optic borescopes, mirrors.

#### 5.1.2 REFERENCE STANDARD

Not required.

#### 5.1.3 INSPECTION TECHNIQUE

With appropriate lighting and optical aids, inspect for abnormal surface conditions: bulges, dents, scratches, cracks, edge delaminations, corrosion products, wear, etc.

Abnormal conditions should be evaluated to determine if a defective condition exists. Other inspection methods detailed herein should be used as needed. Record and report probable defects in accordance with established procedures.



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## 5.2 TAP TEST (COIN TAP)

### 5.2.1 PRINCIPLE

Surface of part is tapped by hand using a blunt, hard object (often a coin) and tone difference or deadening as compared to surrounding area indicates a delaminated, disbonded, or damaged area.

### 5.2.2 EQUIPMENT

Any lightweight, hard-surface object that will not mar the part surface. Electrical/electronic tapping instruments are excluded (see para. 5.7.1).

### 5.2.3 REFERENCE STANDARD

Recommended (currently, this method is usually used without a reference standard).

### 5.2.4 INSPECTION TECHNIQUE

Tap, with appropriate object, over inspection area and note any tone changes or sound deadening as compared to surrounding area of identical underlying structure and material gages. Tone change will indicate possible delamination, disbond, or impact damage. It also may indicate micro-cracking and softening in the epoxy matrix.

Tap testing is not reliable and becomes insensitive to delamination and disbonds under certain structural conditions, particularly increasing skin thickness. Optimum sensitivity is associated with very thin skin structure. For this reason, a reference standard containing a defect in a reproduction of the actual part or section of the part should be used for

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validation of the tap test. Failure to positively detect the reference defect disqualifies the tap test method, and other NDT methods detailed herein must be used.

Defect indications should be evaluated and probable defects recorded and reported per established procedures.

### 5.3 ULTRASONIC PULSE-ECHO

#### 5.3.1 PRINCIPLE

Ultrasonic pulses are generated by a single transducer in contact, through a couplant, with the part surface. The pulses travel internally in the part and reflect or echo from each material change, for example, the interface of skin bonded to substructure. The reflected pulses are detected by the transducer and resultant signals on a Cathode Ray Tube (CRT) are monitored for changes caused by defects within the part.

#### 5.3.2 EQUIPMENT

- o Standard ultrasonic test instrument equipped with a high-resolution pulser-receiver unit for 5, 10, and 15 MHz transducers.
- o One each 5, 10, and 15 MHz high-resolution contact search units.
- o One each 5, 10, and 15 MHz transducers adaptable to angle beam shoes.
- o Angle beam shoes.
- o Couplant: oil, grease, commercial couplant. Also, water bubblers, squirters, and encapsulated-water standoffs have been used. The latter is illustrated in figure 1.

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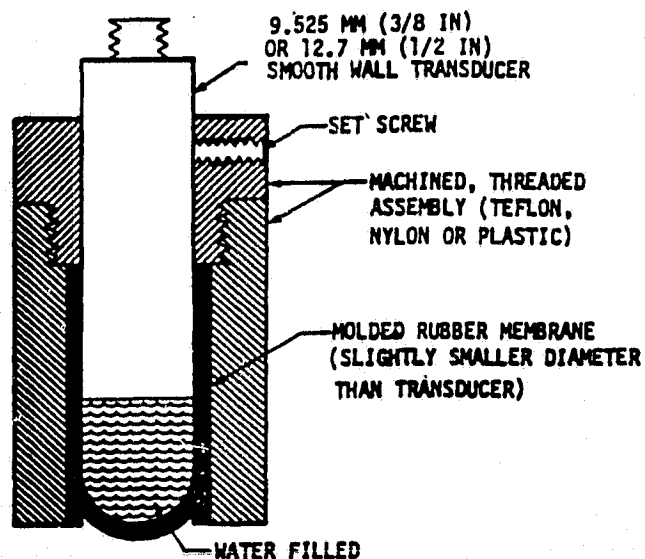


FIGURE 1. STAND-OFF SEARCH UNIT

### 5.3.3 REFERENCE STANDARD

Required (see sec. 6.0).

### 5.3.4 INSPECTION TECHNIQUE

Two methods are described--the straight beam method and the angle beam method. Both are effective in detecting delaminations in the laminated composite skin and delaminations or disbonds at the skin-to-underlying structure interface. The angle beam method is applicable only to metal honeycomb. (It is unlikely that nonmetallic honeycomb will support sufficient sound transmission levels.) The straight beam method is applicable both to skin-to-honeycomb and skin-to-underlying laminate bonds. The two methods are illustrated in figures 2 and 3.

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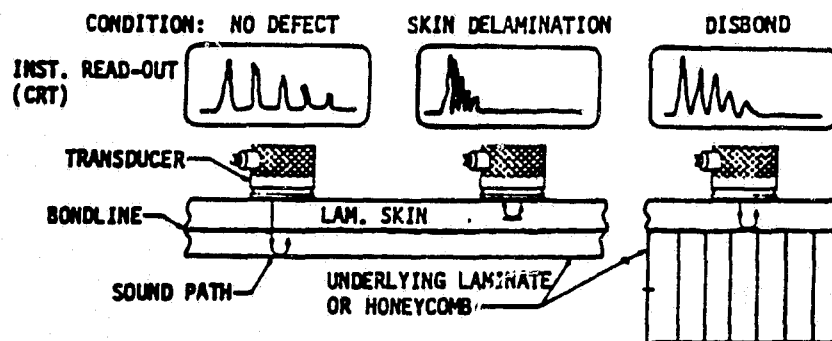


FIGURE 2. STRAIGHT BEAM TECHNIQUE

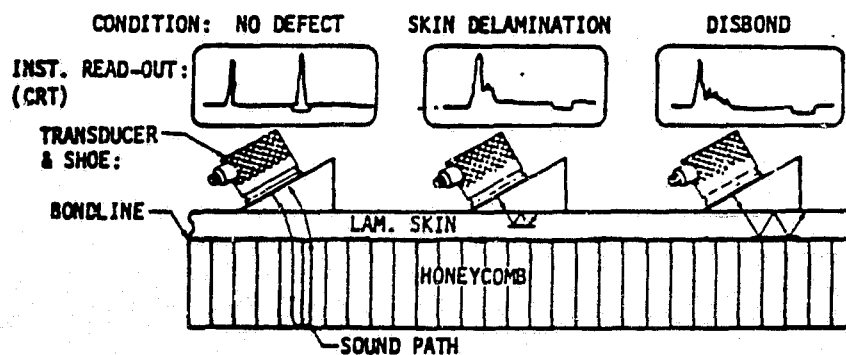


FIGURE 3. ANGLE BEAM TECHNIQUE

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Using applicable instrument setup instructions and reference standards, verify instrument sensitivity with reference standard defects. It may be necessary to vary search unit assemblies using plastic shoes or encapsulated water standoff devices and various shoe angles for the angle beam method. Only certain frequencies or specific transducers may be usable on the structure and defect combination under investigation.

Clean part if oily or dirty, apply couplant, and inspect by moving or positioning search unit over areas requiring inspection. Evaluate defect indications and record and report probable defects in accordance with established procedures.

#### 5.4 ULTRASONIC THROUGH-TRANSMISSION

##### 5.4.1 PRINCIPLE

Ultrasonic pulses are generated by a "transmit" transducer coupled by water bath, water column to the part surface, or by direct contact with the part surface. The ultrasonic pulses enter the part and travel internally through the part and are detected by a "receive" transducer on the opposite side of the part also coupled to the part in the same manner as the "transmit" transducer. Detectable defects are those that block or reduce sound transmission seen as a loss or reduction in signal amplitude on the instrument readout (see fig. 4).

The through-transmission technique is predominantly used to inspect parts prior to installation on an airplane or after removal from an in-service airplane. With the part removed, it can be submerged in water or placed in a holding device for inspection by water column search units. With this setup, transducers do not contact the part surface and mechanical scanning with "C" scan recording is usually performed over the entire part. Defects are detected as anomalies on the "C" scan recording. However, this technique can also be used on installed parts using one of two methods.

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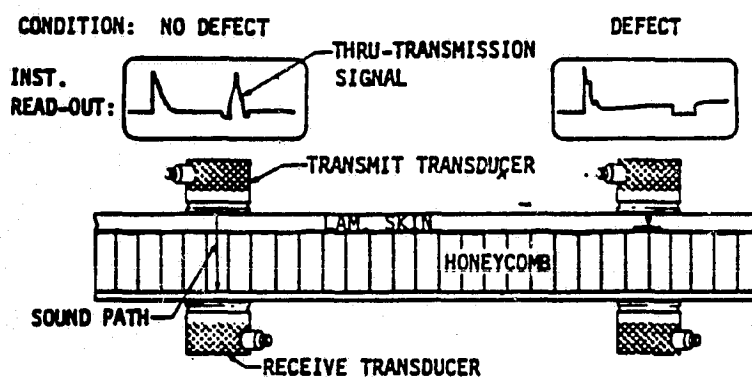


FIGURE 4. THRU-TRANSMISSION TECHNIQUE

One of these is to manually place one of the transducers in direct contact with the part (using a couplant) and position the other transducer over the opposite surface to obtain a maximum through-transmission signal. This is repeated until the area of interest has been covered. The other method uses a yoke assembly to hold both transducers in constant alignment while scanning the transducers over the area of interest. These methods have been used to inspect limited areas of a given part.

#### 5.4.2 EQUIPMENT

- o Standard ultrasonic test instrument equipped with transmit-receive connections for through-transmission operation with 1.0, 2.25, and 5.0 MHz transducers.



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- o Two each 1.0, 2.25, and 5.0 MHz transducers for contact inspection or search unit assemblies for noncontact mechanical scan inspection.
- o If noncontact mechanical scan inspection is used, additional equipment needed includes tank or squirter assemblies, part holding devices, mechanical scan apparatus, and "C" scan recorder coupled to the mechanical scanner and connected to the "recorder" output of the ultrasonic instrument.
- o Oil, grease, or commercial couplant.
- o Yoke (optional): Yokes are built for specific application depending on part thickness and required "throat" length, i.e., distance from part edge to inspection area. A yoke design is shown in figure 5.

#### 5.4.3 REFERENCE STANDARD

Required (see sec. 6.0).

#### 5.4.4 INSPECTION TECHNIQUE--CONTACT METHOD

Set up instrument per applicable instructions, place transducers on reference standard, and identify through-transmission signal. Verify signal and sensitivity by positioning over defect location and observing signal loss. Select best frequency and transducers for sound transmission through the part and optimum defect sensitivity. The lowest frequency is usually best for high-attenuation materials. Defect resolution is best at the higher frequencies.

Clean part if greasy or dirty and inspect by the contact method described in section 5.4.1. Evaluate defect indications and record and report probable defects in accordance with established procedures.

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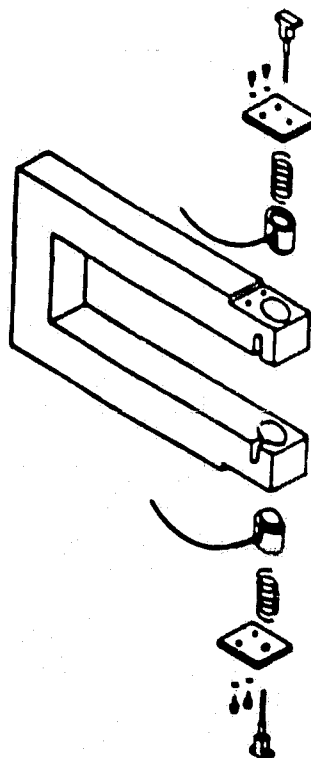


FIGURE 5. YOKE FOR THRU-TRANSMISSION INSPECTION \* \*(FROM "S-3A GRAPHITE/EPOXY SPOILER DEVELOPMENT PROGRAM, VOL II" JULY 1975, E.G.BLOSSER, ET AL.)

#### 5.4.4 INSPECTION TECHNIQUE--WATER IMMERSION OR WATER COLUMN

Follow equipment manufacturer instructions and internally developed procedures for mechanical scan/"C" scan recording inspections in a water bath or when using water column search units.

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## 5.5 ULTRASONIC DIGITAL THICKNESS GAGE

### 5.5.1 PRINCIPLE

This method uses the pulse-echo principle wherein a transducer in contact, through a couplant, with the part surface generates a pulse that travels to the opposite surface of the laminated skin, echoes from this surface, and returns to the transducer. The time interval from pulse initiation to its return to the transducer is measured electronically and presented as the part thickness on a digital readout. This method detects delaminations in a laminate and may detect disbonds if the skin-to-underlying structure bond permits the pulse to pass into the underlying structure. It is illustrated in figure 6.

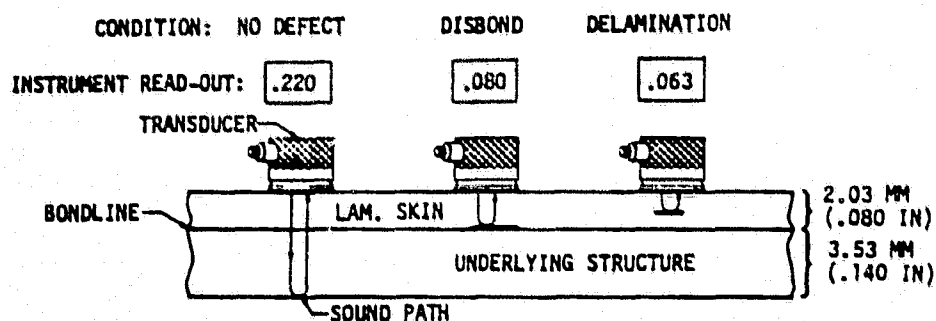


FIGURE 6. DIGITAL THICKNESS GAGE METHOD

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#### 5.5.2 EQUIPMENT

Digital readout ultrasonic thickness test instrument and recommended transducers. Oil or commercial couplant.

#### 5.3.3 REFERENCE STANDARD

Required (see sec. 6.0).

#### 5.5.4 INSPECTION TECHNIQUE

Using the applicable instrument setup instructions and reference standard, calibrate instrument on a thickness reference standard to verify proper instrument adjustment and sensitivity throughout the thickness range of materials to be inspected. Clean part as necessary. Inspect by applying couplant and placing the transducer on the surface for each reading. Reliable readings cannot be obtained when sliding the transducer along the surface for a scan type of inspection.

Defect indications will appear as thinner than normal readings characterized by an abrupt change in thickness reading. Some variation in thickness reading of total thickness may result from local variations in resin content. However, these should not be abrupt and would appear as a variable total thickness reading.

Evaluate all defect indications and record and report probable defects in accordance with established procedures.

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## 5.6 RADIOGRAPHY

### 5.6.1 PRINCIPLE

A beam of penetrating radiation in the form of X or gamma rays is directed through a part or structure with such energy as to be partially absorbed. Sensitized film is placed on the opposite side of the test object. After exposure and processing the film, the transmitted portion of the beam will have formed a density image of the test object on the film. The image density is proportional to the variation of absorption by the test object.

Because X-rays are electrically generated, the radiation energy (penetrating power) can be selected to any appropriate absorption-transmission ratio for a wide range of test objects. Isotope-generated gamma rays are emitted at fixed energies and thus are limited to the radiography of specific material thicknesses.

### 5.6.2 EQUIPMENT

Iridium 192 (300 to 600 kv) and cobalt 60 (1200-1300 kv) are two isotopes in wide use for metal radiography. With steel penetration of 2 inches and 8 inches respectively, they are obviously unsuitable (too powerful) for the general metal airframe components requiring 50 to 250 kv. X-ray machines rather than isotopes are used in the 50-250 kv range and are in common use as standard aircraft inspection equipment.

With the introduction of nonmetals into aircraft structures, lower kv equipment is becoming necessary. Energy levels as low as 10 kv are suitable for graphite-epoxy thicknesses. X-ray machines with beryllium windows and 10-50 kv are recommended for low-kv inspection of advanced composites.

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#### 5.6.3. REFERENCE STANDARD

Not required. Sensitivity is assured by using the optimum radiographic technique developed for the structure and defect type being inspected.

#### 5.6.4 LOW-KV RADIOGRAPHIC TECHNIQUE

Normal radiographic practices are followed except that, for aluminum or graphite thicknesses that are less than 0.10 inch, the X-ray tube distance from the film may be minimized without loss of image sharpness. This plus the use of a medium-speed film permits the lowest possible exposure kilovoltage for a reasonable exposure time.

### 5.7 BOND TEST INSTRUMENTS

#### 5.7.1 PRINCIPLE

Several instruments of different types and principles of operation are grouped under this section. They are used primarily as adhesive bond test instruments. However, they are also applicable to composites for detection of ply delamination, disbonds, impact damage, and possibly others. Specific capability depends on the instrument used, the specific structure, and the defect type. Operating principles are summarized as follows, although the manufacturers' instrument manuals should be consulted for detailed information.

Sound generation is accomplished by piezoelectric transducers, electromechanical tapping, or induced eddy currents causing vibration of metal (conductive) members in the structure. Some instruments use fixed frequencies, some a variable frequency as selected by the operator, and some a repetitive sweep of frequencies through a given range. Frequencies used

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may range from a few kHz to as high as 80-100 kHz. All instruments require transducer contact with the surface, but some do not require a couplant. Detection of the sonic energy, as modified by the structure and possible defects, is accomplished by piezoelectric transducers or microphone devices. Readout methods include cathode ray tube, meter, and audible or visible alarms.

#### **5.7.2 EQUIPMENT**

- o Any of several bond test instruments currently being marketed that will clearly detect the simulated defects in the required reference standards.
- o Search units and other accessories required.
- o Some instruments will require the use of a couplant--oil, grease, or commercial couplant.

#### **5.7.3 REFERENCE STANDARD**

Required (see sec. 6.0).

#### **5.7.4 INSPECTION TECHNIQUE**

Because the sensitivity of these instruments varies considerably with the type and geometry of the structure, it is important that the exact structure with appropriate defects (reference standard) be used in initial instrument setup and sensitivity verification. Set up the instrument and verify sensitivity in accordance with established procedures. Clean part as needed, apply couplant if required, and inspect by placing the search unit at specific locations throughout the area to be inspected. Sliding the transducer over the surface in a scanning motion may be permissible, but should be verified as a reliable procedure on the reference standard.

Evaluate all defect indications and record and report probable defects in accordance with established procedures.

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## 6.0 REFERENCE STANDARDS

Reference standards are required for in-service inspection of advanced composite structures with ultrasonic and bond tester methods and recommended for tap test. The best reference standard would be a duplicate part or section of structure to be inspected, including the actual defects of concern. However, obtaining parts with defects of the desired sizes and locations is not always possible, and the alternative is to fabricate reference standards that duplicate the parts to be inspected with built-in simulated defects of the required sizes and locations.

An alternative to the built-in defect standards is the step wedge thickness standard, which contains several thicknesses each representing a delamination or disbond depth below the surface. This type of reference standard simulates large defects only. It is suitable for ultrasonic pulse echo, ultrasonic thickness gage, and bond tester standardization. It is not suitable for tap test validation.

The following reference standards are recommended to be consistent with the most common nonvisual defects occurring in graphite-epoxy composite structures and as dictated by nondestructive test methods standardization requirements.

### 6.1 DELAMINATION/DISBOND STANDARDS

These are suitable for standardization of ultrasonic pulse-echo and through-transmission, bond test, and validation of tap test methods. They are fabricated by placing Teflon tape, mylar-covered porous filter material, Tedlar, precured adhesive, or prepreg tape in the bondline and between plies of a laminated part. The size is determined by the defect allowables criteria and shape may be either round or square. Figure 7 is typical. Step wedge thickness standards are illustrated in figure 8.



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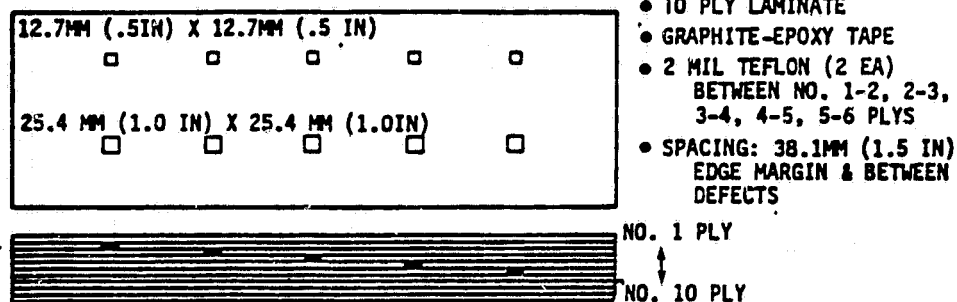


FIGURE 7. TYPICAL GRAPHITE-EPOXY DELAMINATION AND DISBOND REFERENCE STANDARD

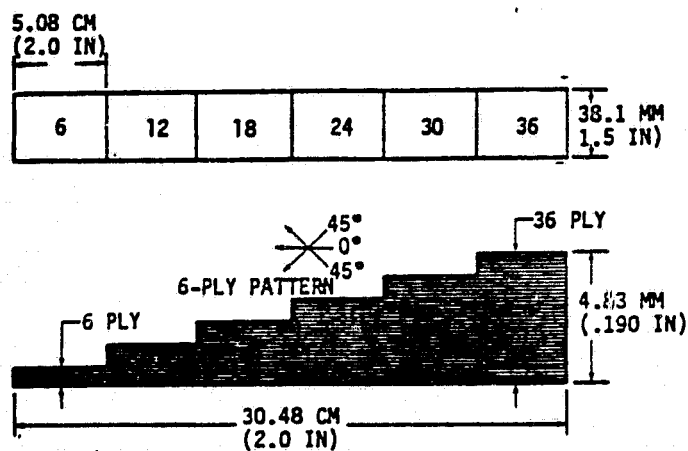


FIGURE 8. STEP-WEDGE THICKNESS REFERENCE STANDARD

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## 6.2 IMPACT DAMAGE AND ENTRAPPED WATER

While reference or "verification" standards are not commonly used for these defects, sensitivity assurance and confidence levels would be enhanced with their use.

**Impact Damage:** This is created in the laminated skin member by impact of a blunt object with the test standard surface. A typical method is illustrated in figure 9. Impact defects in the reference specimen should range in severity up to visible damage. The nonvisible defects are used to verify sensitivity of ultrasonic, bond tester, and tap test methods.

**Entrapped Water:** This reference standard can be used to verify the capability of radiography and other methods to detect water-in-honeycomb structure. A tiny drilled hole through the skin in the center of one or more cells allows introduction of water with a hypodermic needle or similar device. Water amounts can be varied to determine detectable levels for various methods.

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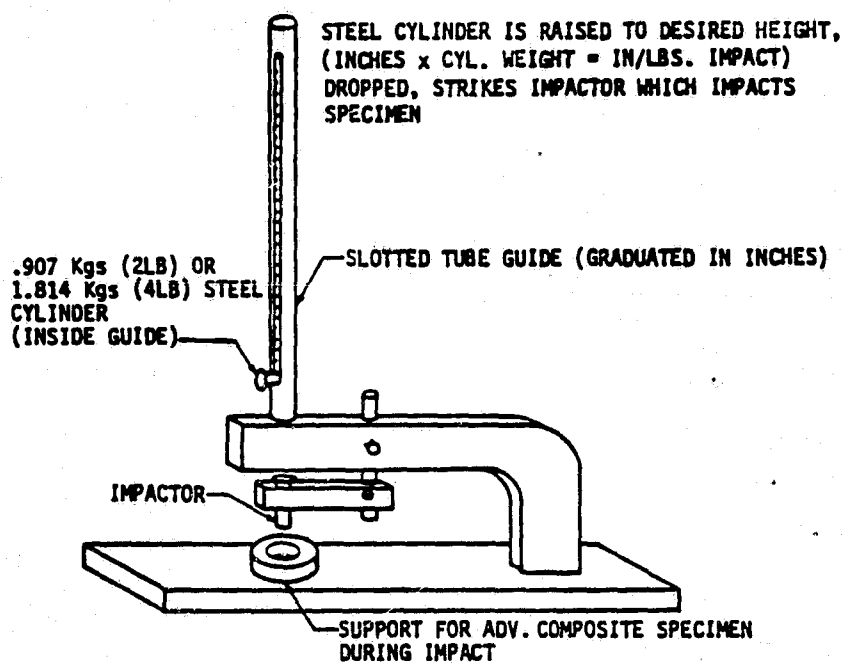


FIGURE 9. METHOD TO ACCOMPLISH CONTROLLED IMPACT DAMAGE

Literature ReviewORIGINAL PAGE IS  
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Literature on bearing strength appears to be most useful in designing mechanically fastened joints against failure in the bearing mode. As such, discussion of bearing strength in the literature is generally found as a subtopic of joint strength. The reader is therefore referred to the literature in the "Joint Strength" section of this report. In that literature, occasional mention of bearing strength will be found.

A review of early literature concerning bearing strength is available (Padawer 1972), with emphasis on improving strength of laminates with bolt holes up to 200% by colaminating thin sheets of Boron or steel.

Studies of the effects of hygrothermal excitations on bolt or pin bearing strength are available (Wilkins 1977, Kim and Whitney 1976). Gr/E bolt bearing strength is reportedly (Wilkins 1977) degraded a maximum of 18% by the environment.

Other efforts include studies of stacking sequence effects in pin-bearing Gl/E (Quinn and Matthews 1977), failure mode and strength predictions for anisotropic bolt bearing specimens (Waszczak and Cruse 1971), and stress analysis around pin loaded holes (deJong 1977).

## Bending Strength

A small amount of literature treats bending strength. Many aerospace designers report that bending strength data is of limited use, because bending loads acting on built-up structure are typically transformed into tensile, compressive, and shear loads. For example, an entire wing structure supporting an airplane must have bending strength, however, the load is usually spoken of in terms of compression on the upper wing skin and tension on the lower skin. A possible exception comes to mind, wherein bending moments, and other loads, are applied to a solid composite beam not having built-up substructure. This is the YUH-60A helicopter trail rotor carry-through beam. Analysis of plate bending and comparison to Gr/E data shows that some difficulty may be encountered in interpreting flex data (Whitney, et al. 1974). Bending stiffness is shown to depend upon stacking sequence and other design and manufacturing variables. The authors suggest that flex data may be of limited utility. What has received greater attention than bending strength in the literature is bending stiffness, and the reader is referred to the appropriate literature on lamination theory (Jones 1975, Ashton, et al. 1969).

The transformation of applied bending loads into other types of loads is evident both in literature concerning structural design (Ashton, et al. 1972) and in literature concerning test methodologies (Berg, et al. 1972, Hill and Anderson 1972). For example, ASTM D2344-67 Test for Apparent Horizontal Shear Strength of Reinforced Plastics by Short Beam Method is shown (Berg, et al. 1972) by analysis to underestimate maximum shear stress in unidirectional Gr/E by upwards of 100%. The authors suggest using the test only for material screening purposes. The ASTM bending test is shown to be inadequate for laminate stiffness and strength characterization purposes (Jones 1976a). Shear deformation and lamination effects have been included in analysis of the flexure test (Pagano 1967). The three point bending test has been used, with stress analysis, to characterize K49/E (Zweben 1978), and also to study creep phenomenon (Tang 1970).

Some data is available concerning hygrothermal effects on bending strength. Worldwide outdoor long-term exposure tests of Gr/E, Gr/Polysulfane, and K49/E are being conducted by NASA Langley Research Center (Pride 1978), and flexure strength is reportedly not degraded after 3 years exposure. Elevated temperatures to 300°F are shown (Browning, et al. 1977) to significantly degrade bending strength of Gr/E in the presence of moisture, whereas very little degradation occurs in a dry environment. Some insight concerning the effects on shifting material properties and failure modes is offered. The same data appears to be available in another publication (Whitney 1976). Elevated moisture and temperature bending strengths of several other materials including Gr, B, S-GI, K49, and neat resin have been reported (Browning and Hartness 1974). Another author (Broutman 1976) reports that unidirectional Gr/E bending strength is rather insensitive to long term, 11 years, aging and moisture exposure. Salt-water excitation was also studied. Bending strength of B/Al has been studied with simulated saltwater excitation (Dardi and Kreider 1974).

Thermal cycling with vacuum is reported (Berman 1975) to have no effect on bending strength of unidirectional Gr/E. Bending strength of a composite less sensitive to elevated temperatures is reported (Browning and Marshall 1970).

Torsional load cycling is shown (Sumsion and Williams 1975) to degrade residual bending strength and stiffness of Gr/E.

B/E bending strength and stiffness as a function of radiation exposure has been reported (Bullock 1974).

The effects of some design and manufacturing variables on bending strength have been studied. Thermoplastic matrix material has been combined with Gr and the composite properties including bending strength evaluated (Maximovich 1977). Such matrix materials offer potential composite cost and processing advantages, including avoidance of exotherm problems in thick laminates. Recent communications (with Chase, Daugherty, and Doyal) indicate that greatly improved thermo-

plastic materials are becoming available. Bending strength and other properties of short fiber reinforced composite are reported (Knight and Hahn 1975). Word-of-mouth reports indicate that great improvements in whisker-reinforced composites are in progress. Prestressing of prepregs either in bending or tension has been shown (Mills, et al. 1975) to offer potential of 10% average strength increase, and 50% reduction in standard deviation. Improved bending and shear strengths are reported possible by neutron-irradiating carbon fibers (McKague 1973). The ratio of  $G_I$  to  $G_r$  in a hybrid has been shown (Kalnin 1972) to effect the bending strength and other properties of the hybrid. Dependence of bending strength of 3D  $G_I/E$  composite upon fiber aspect ratio, volume ratio, and surface finish has been measured (Suezawa 1977).

Studies of failure modes of C/C in bending are reported (Frye and Rayner 1972).

Compressive StrengthORIGINAL PAGE IS  
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Compressive strength literature is less common than tensile strength literature. The available literature indicates that different compressive failure modes can be forced by varying test apparatus support conditions (Ryder and Black 1977). As a result, there is debate concerning appropriate specimen geometry and test apparatus. Such discussion considers, for example, whether compressive strength should be measured with or without side supports or edge supports, what the supports should be, if used, specimen length, width, and thickness. As with many composite material tests, no standard methods have gained universal acceptance. Until such standardization occurs, it is, therefore, most important to report particular test conditions and failure modes along with numerical data.

Compressive strength has been shown to depend upon design and manufacturing variables (Greszczuk 1974). One paper raises the possibility of a coupon size effect in compression testing (Ryder and Walker 1977). Many papers report dependency upon fiber volume ratio, and one (Kalnin 1972) reports the dependency upon G1/Gr hybrid ratios. G1 Whisker/E compressive strength is shown (Suezawa, et al. 1977) to depend upon many variables, including aspect ratios, volume ratio, and surface finish.

Compressive strength of Gr/E in the presence of a hole is shown to be predictable using a fracture mechanics-derived technology (Walter, et al. 1977). For details of LEFM of modeling, refer to "Fracture Resistance" and "Notch Sensitivity" sections of this report.

An earlier in-depth review of compressive strength literature was completed in 1973 as part of a Ph.D. thesis (reference [1] in Davis 1975), and is recommended as a starting point. In another paper (Davis 1975), the same author presents a micromechanics analytical model of compressive strength, and compares the results



to data on unidirectional B/E. The model accounts for initial fiber curvature and considers two possible modes of failure; delamination and shear instability. The later mode is shown to dominate. Reviews and references to micromechanics compressive strength literature are available in Chapter 3.5.3 of Jones 1975). Jones provides an excellent overview of much of the analytical and experimental literature (Ashton and Love 1969, Daniels 1970, Lager and June 1969, Chung and Testa 1969, Hermann, et al. 1967, Hanasaki and Hasegawa 1974, deFarran and Harris 1970, Lanir and Fung 1972, Lovell 1972). More recent buckling and compression analysis is evident in the literature (Whitney and Sun 1975, Crouzet-Pascal 1978, Wang, A.S.D. 1978, and Simites and Giri 1977). One paper (Spier and Wang 1975) offers a note of caution to designers concerning unidirectional E/A1.

More recently, compression test procedures have been reviewed and considered (Ryder and Black 1977) to fall into two categories:

- (1) End loaded coupons
- (2) Sandwich beams.

An experimental apparatus reported to produce a pure moment has also been used to measure compressive strength (Hill and Anderson 1972).

Ryder and Black describe the failure modes peculiar to the first specimen type, and delineate problems associated with the test method. A modified end loading apparatus and experiment on large gauge length Gr/E is then reported, and stress-strain curves are given.

The literature is generally concerned with in-plane compression loading of the laminate, although at least one or two tests (Fahmy, et al. 1976, Unknown NBS) report data in what appear to be the thickness or transverse direction.

Some of the literature deals with compression effects on fatigue strength and the reader is referred to the "Fatigue Strength" section of this report. For convenience, these papers are listed here (Sims and Brogdon 1977, Pillar, et al. 1977, Kunz and Beaumont 1975, Porter 1977, Kasen, et al. 1977, Sendekyj, et al. 1977, Roderick and Witcomb 1977, Ramani and Williams 1977, Ryder and

Data for B/A1 is available showing non-linear "hardening" effects (Herakovic', et al. 1977).

Studies of low velocity impact effects on compressive strength of thin laminates have shown that representative impacts of runway debris and perhaps hail can cause 40 to 60% degradation (Hardrath 1977). Note that a wing upper surface is loaded primarily in compression and may be subject to dropped tools and hail. The same author reports that the tests, using 13 mm diameter balls, show strength loss of the same order as fastener holes, and that such considerations are likely for a while to limit allowable strain to .005 in civil aircraft. Hardrath also reports:

Studies have been made of the effects of low-velocity impact damage on compressive strength of hat-stiffened panels. The strength of panels designed to resist buckling under modest loading intensity (.0034) were found not to be affected significantly by the impact. However, the strengths of panels designed not to buckle below .008 strain were damaged severely enough to reduce their buckling strain to approximately .004, a value slightly lower than that attained for a panel containing a 13-mm diameter hole.

References pertinent to the previous discussion may be found in Hardrath's paper, along with several listed here (Rhodes, et al. 1978, Rhodes, et al. 1976, Walter, et al. 1977).

Buckling has received attention in the literature from several standpoints. It is reported (Spier and Klouman 1977) that compressive strength of laminates, especially unidirectional laminates, is not predicted accurately using classical orthotropic elastic buckling theory. The authors argue that local crippling strength of stiffener elements is needed to perform an appropriate short column analysis (ref. [3] in Spier and Klouman 1977), and present empirical nondimensional crippling curves for Gr/E no- and one-free edge support conditions. Literature on buckling and compression analysis is reviewed and referenced in Chapter 5 of Jones 1975.

The effects of bow-type imperfections on the mass and buckling load of Gr/E blade-stiffened panels have been studied on the computer (Stroud, et al. 1977). Initial bow has been shown to reduce allowable buckling load by up to 47%. At NASA Langley Research Center, a great deal of work on computerized methods for design of optimum lightweight panels for compression is available (Stroud and Arganoff 1976, Stroud, et al. 1977, Anderson and Stroud 1978, Starnes and Haftka 1978, Williams and Stein 1976, Williams and Mikulas 1976). The studies generally show potential weight savings of 30-50% over efficient aluminum structure. Blades, J's, and hat stiffened structures are treated using various linear and non-linear analysis. Most of these papers also report comparisons to Gr/E experimental data. Much improvement over earlier design methods (Bush 1972) is evident. Various computerized analyses of compression are reviewed and shown to be deficient (Weller 1977a) in predicting failure mode and strength of Gr/E and B/E. The same author, in another paper (Weller 1977b) compares two types of specimens of Gr/E and B/E, and finds coupons to be preferable to cylinders. Many data curves are presented.

Apparently unpublished studies of ongoing research by Lauritis of Lockheed-California and Whitney of USAF concerning hygrothermal effects on Gr/E compressive strength were reported at the Bergamo Center Conference. Long term outdoor exposure of coupons is reported (Pride 1978) to have little effect on Gr/E compressive strength. Hygrothermal effects on buckling moduli are reported (Shen and Springer 1977) along with some pertinent references.

The effects of strain rate, voids, and fillers on epoxy composite compressive strength have been reported (Ishai and Cohen 1969). Compressive strength with an edge notch in Gl/E has been reported (Vanderveldt and Liebowitz 1968). Thermal spike effects on Gr/E are reported in the literature (Mazzio and Mehan 1977) and the reader is directed to the plotted results. Cryogenic compressive properties for common composites are reported and generally appear quite good (Unknown, NBS).

Several papers report development of structure with consideration of compressive strength (Elliott 1977, Ashton, et al. 1972). Perusal of the "Aerospace Applications" section of this report will indicate that quite a bit of compressive structure is in service, however, there is a noteworthy absence of fuselage structure. It is reported (personal communications with designers) that many composite fuselages would have to enter the post-buckling range to be competitive with aluminum structure. Lack of extensive knowledge concerning long term compressive, buckling, and fatigue properties in the service environment are seen as major contributing impedances to fuselage development.

## Density

Low density is one of the inherent advantages of composites. It is relatively easy to quantify and predict. Hence this material property receives limited treatment in the literature. In general, one can find in many documents the following typical densities of composite constituents:

	Density		Specific Gravity
	Kg/m <sup>3</sup>	lb/in <sup>3</sup>	
Epoxy	1200	.04	1.2
Aluminum	2800	.10	2.8
Carbon, Graphite	1700	.05	1.7
Glass	2600	.09	2.6
Polyester	1400	.05	1.4
Boron	2600	.09	2.6

and for purposes of comparison:

Titanium	4500	.17	4.5
Steel	7800	.28	7.8

(adapted from Jones 1975 and Tsai and Hahn 1977)

A rule-of-mixtures expression (Tsai and Hahn 1977) can then be used if volume fractions are known, and can be modified to account for voids.

Because epoxy is hygroscopic, the total weight of a composite laminate can change when exposed to humid air or, in general, to a hygrothermal environment (Browning, et al. 1977, Browning and Hartness 1974, Springer 1976, Broutman 1976, Freeman and Campbell 1972, Whitney 1976, Hofer et al. 1977). This weight change is typically accompanied by a volume change (See "Dimensional Stability" section of this report), such that a designer usually finds it more convenient to deal with the effect in terms of % change in total weight rather than change in density.

A typical weight increase for a Gr/E aircraft structure reaches slightly greater than 1% (Tenny and Unnam 1978). Analysis is available which can fairly accurately predict the weight increase and variations due to moisture absorption and desorption (Tenney and Unnam 1978, Tompkins, et al. 1978, Unnam, et al. 197x).

### Dimensional Stability

A moderate amount of both analytical and experimental literature is available concerning dimensional stability. Most of this literature focuses upon moisture and temperature effects, with micromechanical models also well represented. An excellent review and comparison of these linear elastic theories with Gr/E experimental data is available (Wang, et al. 1975). It is noteworthy that near-zero values of coefficient of thermal expansion are attainable in Gr/E; a material property which makes this material very attractive in many applications. Also, the sign of K49 coefficient of thermal expansion demonstrates behavior opposite that normally demonstrated by metals (Kasen 1975). Unlike metals, however, a typical Gr/E laminate exposed to water over a long period of time may absorb close to 2% laminate weight in moisture. This 2% moisture absorption can cause a 1% change in linear dimension (Whitney 1976). There is clear evidence (Whitney 1976) that a typical neat (epoxy) resin can absorb around 5% of its weight in water. In addition, most commonly used fibers, perhaps with the exception (Stratton and Mazzio 197x) of Kevlar\*, do not absorb significant amounts of water. With this in mind, an additive volume approach (Shirrell and Halpin 1977) is available which describes the volume increases and decreases accompanying moisture absorption and desorption in a laminate. The moisture-size effect is largely reversible (Freeman and Campbell 1972), although microcracks and some size changes have been reported (Berman 1975, Whitney 1976).

The process of moisture absorption and desorption has been described using models based upon Fick's Laws (Shirrell and Halpin 1977, Browning, et al. 1977, Whitney 1976). A diffusion model modified to account for daily sunlight and humidity variations has also been developed (Tenney and Unnam 1978, Tompkins, et al. 1978) and computerized (Unnam, et al. 197x).

\*Dupont Registered Trademark.

One author (Berman 1975) shows to some extent that moisture has no effect on Gr/E coefficient of thermal expansion. The separate and to some extent combined effects of temperature and vacuum, both cyclic and monotonic, have been studied (Freund 1975, Berman 1975). Moisture absorption is shown to increase the thickness of several common composite materials (Browning and Hartness 1974).

Analytical expressions for the coefficients of thermal expansion ( $\alpha$ ) in principal material directions of unidirectional composites are available, based upon micromechanics. Expressions of  $\alpha$  for angle-ply laminates are available based upon lamination theory. A rule-of-thumb expression for quasi-isotropic laminate  $\alpha$  is available in the literature (Robinson, et al. 1974).

It has been shown (Freeman and Campbell 1972) that many Gr fibers have a slightly negative ( $\alpha = -.2 \times 10^{-6}$ ) coefficient of thermal expansion in the fiber direction, and neat epoxy resin has quite a high positive coefficient ( $\alpha = 33 \times 10^{-6}$ ). As might be expected based upon micromechanic stiffness and geometry considerations, the Gr/E laminate  $\alpha$  in the fiber direction is fiber-dominated (nearly zero), and matrix-dominated in the transverse direction (Freeman and Campbell 1972). Micromechanic expressions along with theoretical curves of variation of B/E  $\alpha$  with fiber volume fraction are available (Chamis 1977).

It has been suggested (Wang, A.S.D., et al. 1975) that better agreement between theory and experiment for  $\alpha$ , particularly at higher temperatures, could be obtained by including viscoelastic effects modeled by Schapery 1968 (Ref. [1] in Wang, A.S.D., et al. 1975). One author (Foye 1975) has considered the thermal expansion coefficient to be time-dependent. Temperature dependent analysis is available (Ishikawa, et al. 1978) for a unidirectional composite. Other analytical efforts are available which treat thermal expansion coefficient in the laminate thickness direction (Fahmy and Ragai-Ellozy 1974, Pagano 1974). Analysis of a material containing ellipsoidal inclusions is reported (Wakashima, et al. 1974). Earlier studies of  $\alpha$  are found in the literature (Halpin, et al. 1971, Budiansky 1970).

Thermal expansion studies of a B/Al cylinder have been reported (Wolff and Esehun 1977).

Analytical micromechanical predictions of the thermal expansion coefficients of composite materials having more complex geometrical variations of spatially oriented fibers, such as those used in rocket nose tips, are available (Rosen, et al. 1977) based upon a repeating unit cell concept. The coefficient of thermal expansion of Gr/E is reported (Freeman and Campbell 1972) to be stable during thermal cycling if the peak temperatures are below the boiling point of water.

Very precise measurements of Gr/E dimensional stability under various influences has been reported (Freund 1975). The effects of cryogenic temperatures on the  $\alpha$ 's of G1/E, Gr/E, B/E, and K49/E are reported (Kasen 1975).

Experimentally, some difficulty may be expected in separating moisture and temperature effects. This follows from the evidence that temperature excursions tend to change the moisture content of epoxy (Freeman and Campbell 1976) and the desire to look at temperature effects at constant laminate moisture percentage. Also, the moisture diffusion coefficient of Gr/E is shown to vary with temperature (Whitney 1976). In light of such difficulties, it is noteworthy that Gr/E laminates typically used in aerospace applications exposed to typical aircraft environments are reported (Tenny and Unnam 1978) to reach a somewhat stable moisture content of about 1%.

Frictional wear rates of Graphite composites for use in unlubricated sliding bearings have been reported (Brown and Blackstone 1974).

Frictional wear rates of Glass composites have been reported (Ward 1974).

Effects of design and manufacturing variables on subsonic rain erosion behavior have been reported (Schmitt 1974). Analysis of rain erosion of coated materials is also available (Springer, et al. 1974).



## Electrical Conductivity

The electrical conductivity of Gr/E and particularly of carbon and graphite fibers has recently become of great interest because of the potential hazard that highly conductive and mobile airborne fibers may pose to electrical equipment (refer to "Carbon (Graphite) Fiber Incidents" in "Service Applications" section of this report). Despite the high conductivity of individual graphite fibers, the Gr/E composite laminate often displays poor electrical conductivity because the epoxy matrix insulates the fibers.

It has been found, however, that special interference fit fasteners placed into drilled holes in Gr/E laminates can be used to gain conducting paths to the fibers, and that the laminate will then conduct. Hence the desire to use a Gr/E structure as the electrical "ground" for a circuit may be feasible (personal communication with General Dynamics). Lightning protection requirements of composite aircraft structure have generally resulted in use of conductive metal skins, grids, perimeters, or meshes, and many problems have ensued. Studies on protecting B/E structure against lightning strokes have been summarized and reported (Clark 1974). Some attention is given to optimum weight coatings based on residual strength after simulated lightning strokes.

Despite such design problems, only a small amount of literature is found concerning electrical conductivity as a composite material property. Some literature concerning the fiber hazard study is referenced in the "Aerospace Applications" section of this report under the heading "Carbon (Graphite) Fiber Incidents" and a great deal more is expected to be available in light of recent funding on the topic.

Various efforts at relating Gr fiber electrical and thermal conductivity to microstructure, modulus, and to each other are reported (Kalnin 1975). The data is limited to unidirectional fiber bundles, however, data for more than a dozen types of fibers is presented. Bounds for electrical conductivity have been analytically derived (Sanford and Silnutzer 1971). Thermal conductivity of particulate composites has been treated analytically (Hashin 1968).

### Fatigue Strength

A lot of literature is available concerning fatigue strength, however, the very cautious optimism amongst designers indicates that there are still many uncertainties concerning cyclic load effects. The accumulated service experience certainly has continued to show fatigue failures to be quite rare in composite structure, as was the case in 1977 when Hardrath wrote:

"As of this writing, fatigue failures have been so rare in composite structures that one frequently hears the generalization that advanced composites do not suffer fatigue. However, as cited earlier, present experience is almost exclusively with structures that are stiffness-critical and, thus, enjoy comfortable margins on strength or strain. Further, many of the components are essentially free of discontinuities that are common in wing or fuselage structure. Prudence dictates that this generalization be examined carefully when composites are introduced into primary structures where higher structural efficiencies are likely to be required. Current understanding of the basic failure modes is rather limited and systematic quantitative data are rare."

It is significant to note that since the time Hardrath's commentary was written, more primary structure has been built and flown successfully, and some useful research on fatigue has continued. Pertinent comments on fatigue research are available in the summary statement of STP 636 by Lauritis 1977.

Despite such apparent progress, it seems that fatigue strength is one of the most worrisome properties to composite designers and one of the major impedances to development of more primary structural applications. Most of the literature is quite interesting, however, much of it has been of limited utility to designers until more recently. Indeed, in light of time or frequency effects much of the experimental literature may deserve reconsideration. Fatigue strength of structure has often been assured by full scale or subcomponent testing, for lack of more generally applicable, cheaper, quicker, and easier coupon or test specimen methods. Many structures have been designed to ultimate strength requirements, which have in many cases enveloped fatigue needs (Hadcock 1972).

Considering the state-of-the-art understanding of metal alloys fatigue, one can well imagine the added complexities of composite fatigue. The literature collectively shows that the morphology of composite fatigue may typically be influenced by many excitations, flaws, and design and manufacturing variables. Also, it unfortunately shows that what applies to one composite, say G1/E, cannot directly be used to understand, say Gr/E, because damage may accumulate in a wholly unique fashion. Consequently, the results of a given experiment may be quite useless if a certain critical piece of information is not reported. Add to this the possibilities that some tests may be constant load while others are constant deflection, some measurements may be change in stiffness while others may be tensile strength, compressive strength, or S-N curves, some may be at one frequency while others are at a frequency one order of magnitude different, and you'll find that much data can be generated which is not clearly useful to the designer, or even to the advancement of the art.

There is some remarkably good, pertinent research reported, which we'll discuss.

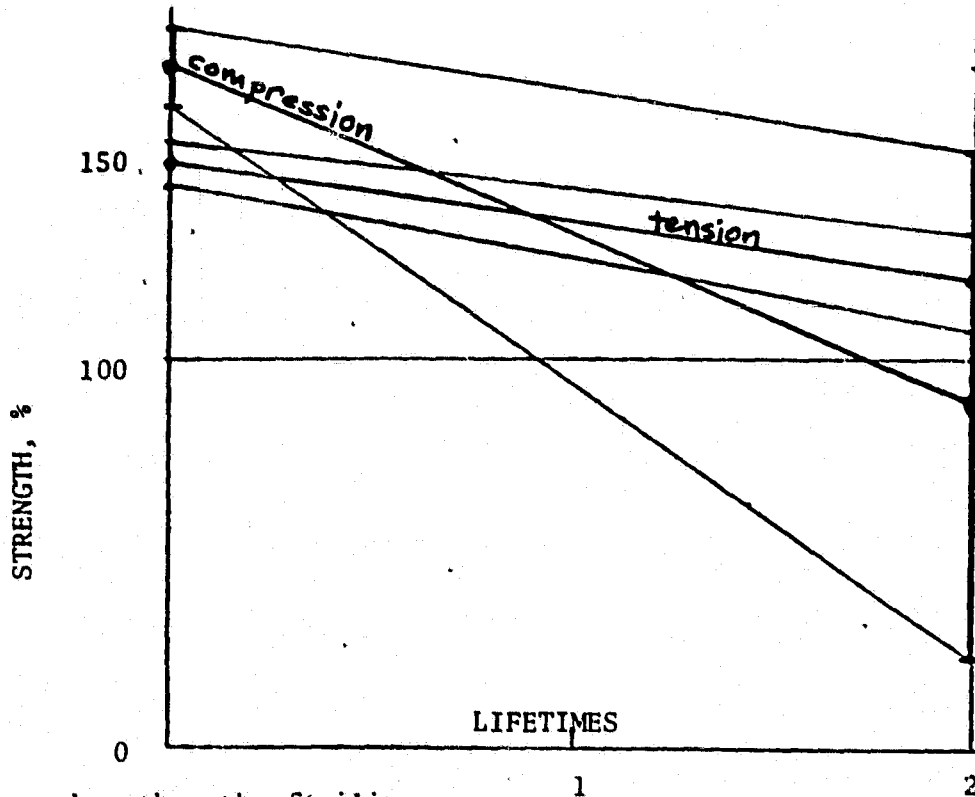
An excellent previous literature review on fatigue is available (Salkind 1972) and is suggested reading as a starting point to newcomers to composite fatigue. Also, a reader interested in roughly comparing composites to metals is directed to such discussions (Grimes 1977, Salkind 1972). ASTM STP 636 is a useful source for fatigue literature.

Many early efforts (Salkind 1972) at characterizing composite fatigue appear to have been derivatives of metals technology. It was noticed that composite damage accumulation could appear in forms different than those typically measured in metals fatigue tests. Hence many composite fatigue tests then measured residual stiffness, strength or whatever was of interest. Discussion of just what

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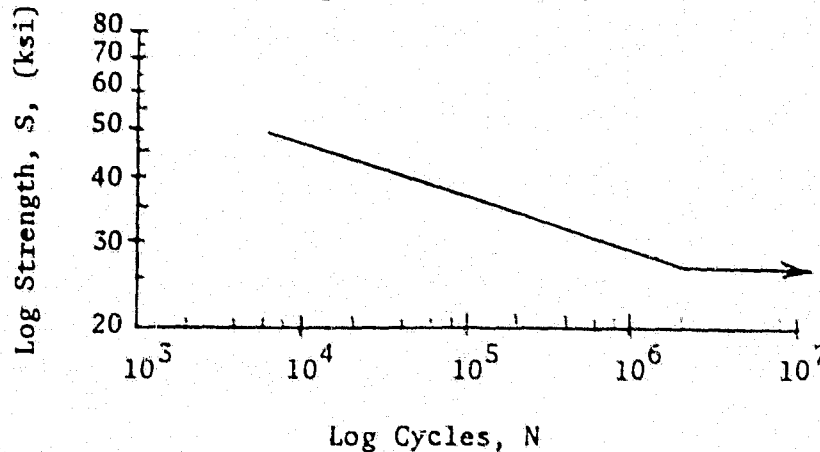
ought to be measured is available (Salkind 1972,1976\*). One can well imagine that in a given structure, several different properties may be of interest, such as residual stiffness, compressive strength, and tensile strength. It appears that the majority of the research community has come to this realization.

Those people more accustomed to metals fatigue may be uncomfortable seeing a plot such as



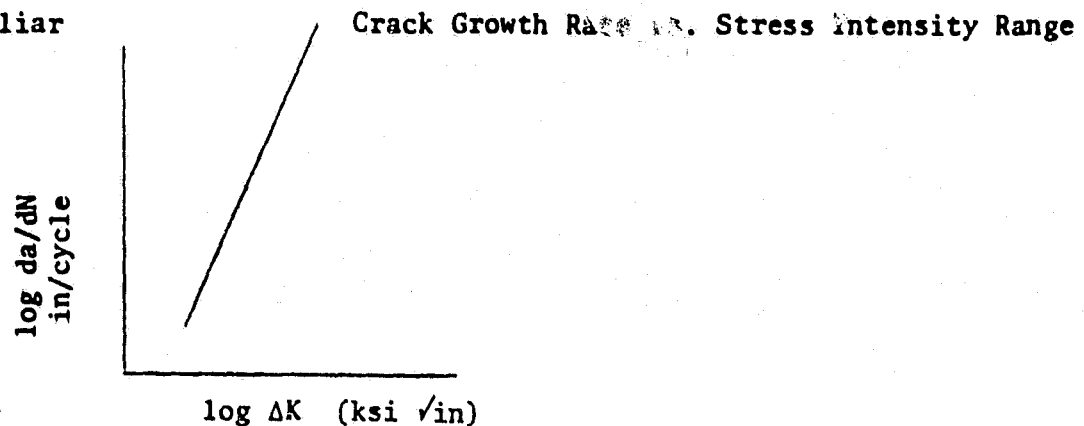
rather than the familiar

S-N Diagram for 1040 Annealed Steel  
(adapted from Shigley 1972)



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or the familiar



The problem with trying to report composite data in the later two traditional forms is that composites in service generally don't accumulate damage in modes consistent with metals. Cracks do sometimes occur and grow, but failure may occur due to damage accumulation in a mode which is not the mode traditionally measured in metals fatigue testing. Despite such problems, one does find S-N curves in the composite literature (Sims and Brogdon 1977, Grimes 1977).

Quite a bit of the literature goes into great detail concerning NDE during fatigue. Some is useful in tracking fatigue damage accumulation and sequences, however, many NDE methodologies reported are constrained to laboratory situations. A few papers are oriented primarily towards NDE monitoring during fatigue (Hamstad and Chiao 1975, Salkind 1976\*, Roderick and Whitcomb 1975, Chang et al 1977, Schwabe 1975) while most others focus upon changes in material properties concurrent with NDE indications,

Much of the fatigue literature studies the effects of excitations, design and manufacturing variables, and/or flaws. Due to the complex morphology of composite fatigue, it is rather difficult to isolate these effects and draw extensive conclusions from the literature (Lauritis 1977).

We shall report the literature within the categories of the effects studied.

#### Materials Studied in Fatigue

First, for those interested in digging up a bunch of fatigue references on a particular composite material, the following citations may be of some use:

for Gr/E (Petrie 1975, Porter 1977, Reed and Eisenmann 1975, Berens and West 1975, Grimes 1977, Chang, et al. 1975, Sims and Brogdon 1977, Francis 1976, Yang 1977, Sendekyj, et al. 1977, Chang, et al. 1977, Ramani and Williams 1977, Papirno 1976, Rosenfeld and Huang 1978, Walter, et al. 1977, Ryder and Walker 1977, Awerbuch and Hahn 1977, Yang 1976, Sendekyj and Stalnaker 1977, Heller et al. 1976, Bhatia 1976, Sumsion and Williams 1975, Kunz and Beaumont 1975, Konishi and Johnston 1978, Bhatia and Verette 1975, Ramkumar, et al. 1978, Salkind 1972, Francis, et al. 1977, Yang 1978, Chou and Croman 1978, Yang and Jones 1978);

for B/E (Hadcock 1972, Grimes 1977, Kulkarni, et al. 1977, Kasen, et al. 1977, Roderick and Whitcomb 1977, Yang 1977, Stinchcomb, et al. 1975, Reifsnider, et al. 1975, Reifsnider, et al. 1977, Bhattia and Verette 1975, Sun and Roderick 1977, Ramkumar, et al. 1978, Salkind 1972), Roderick and Whitcomb 1975, Donat 1970);

for Gl/E (Schwabe 1975, Grimes 1977, Mandell and Meier 1975, Mandell 1975, Sims and Brogdon 1977, Miner, et al. 1975, Yang 1977, Dharan 1975, Gauchel, et al. 1975, Ramkumar, et al. 1978, Salkind 1972, Tobler and Reed 1976, Hahn and Kim 1976, Tanimoto and Amijima 1975, Dally and Broutman 1967);

for B/A1 (Chamis and Sullivan 1975, Tarn, et al. 1975, Dvorak and Tarn 1975, Kendall 1977, Elliott 1977, Kasen, et al. 1977, Carlisle, et al. 1975, Reifsnider, et al. 1975, Reifsnider, et al. 1977, Salkind 1972, Hancock and Shaw 1974);

for K/E (Petrie 1975, Hamstead and Chiao 1975, Grimes 1977, Miner, et al. 1975);

for hybrids (Porter 1977, Hofer, et al. 1977, Grimes 1977, Bhatia and Verette 1975, Salkind 1972);

and for other materials (Pillar, et al. 1977, Salkind 1972, Mandell 1975, Nielsen 1975, Kim and Ebert 1978),

#### Time Effects on Fatigue

The fatigue literature generally reports experiments run at 30 cycles/second. Much of this data may be of limited use because of time at load or frequency effects, in as much as many aircraft structures are expected to accumulate fatigue

loading primarily at about 1 Hz rather than 30 Hz. More specifically, damage measured at 30 Hz may err by one order of magnitude on the non-conservative side when applied to 1 Hz. Recently the time effect has received increased treatment in the literature, and the reader is encouraged to consider all fatigue data in light of such findings. Frequency effects were noticed some time ago (Dally and Broughtman 1967).

Stiffness of B/E and B/Al with holes is shown (Riefsnider, et al. 1975, Riefsnider, et al. 1977, Stinchcomb, et al. 1975) to degrade more at lower frequency than at higher frequency, and the effect is shown to be nonlinear. Various explanations have been considered (Stinchcomb, et al. 1975), however, many of the effects remain unpredictable and unexplained. It is interesting to note that negligible frequency effects from .1 to 10 Hz have been reported on whisker reinforced surgical implant materials (Pillar, et al. 1977) using a different definition of damage. Much of the pertinent research on time effects has been reviewed (Sendeckyj and Stalnaker 1977) in an investigation of a proposed time-at-load model. The model is shown to have applicability to a limited set of data on Gr/E, knowing the failure mode. Some comments and data on time at load effects in Gr/E are also available (Grimes 1977).

A molecular model of fracture in amorphous polymers is available (Valanis and Yilmazer 1978). The model is able to explain some time effects in non-reinforced polymers, and may provide useful insight into polymer composite fatigue time dependency. The same paper reviews other related efforts. It is noteworthy, however, that the metal-matrix composite B/Al exhibits what appear to be time-dependent fatigue effects (Riefsnider, et al. 1975, Riefsnider et al. 1977, Stinchcomb, et al. 1975).

The effects of accelerated testing techniques, particularly in the presence of moisture and temperature, are not clearly understood (personal communication

with Porter and others). Better understanding of such phenomenon will lead to more reliable and efficient structural testing.

### Temperature Effects on Fatigue

Some careful investigations seeking to study the effects of temperature on fatigue strength have been made. The reader is warned that other investigations may inadvertently show temperature influence due to heat generation, particularly at the higher frequencies. Proof of such heating is readily available (Ref. [37-43] in Salkind 1972, Dally and Broughtman 1967).

The nonlinear effects of temperature on residual tensile strength after fatigue loading on a surface notched Gr/E specimen have been studied. An explanation of notched strength based upon a simple mechanics of materials model (Sendekyj, et al. 1977) is available, but is limited to static strength. The combined effects of moisture, temperature, and bending and torsional fatigue upon bending strength and bending stiffness of Gr/E have been studied (Sumsion and Williams 1975), and it appears that moisture and temperature combined may have effects greater than either one alone.

Some qualitative discussion of temperature effects, and of stages of heating during cyclic testing, are available (Dharan 1977) for whisker and continuous fiber reinforced composites.

It is reported that B/E specimen fatigue strength may be improved by heat treatment under load (Sun and Roderick 1977). B/E fatigue at elevated temperatures have been reported using S-N curves (Donat 1970).

Cryogenic effects on B/Al and B/E fatigue properties have been studied (Kasen, et al. 1977). Damage, measured as change in stiffness, is shown to be rather insensitive to temperatures in TT and TC low cycle fatigue. Fatigue strength of SGI/E is shown to improve at 4°K (Tobler and Read).

Fatigue testing of B/Al for purposes of FAA structure approval has been reported (Elliot 1977).



### Moisture Effects on Fatigue

Some literature concerning the effects of moisture on fatigue of advanced composites has recently become available following rather large interest in the moisture effects on G1/E. Water is shown to affect the matrix and the fiber-matrix interface, as well as promoting stress corrosion of G1/E (Gauchel, et al. 1975). Further references on G1/E moisture and fatigue phenomenon along with saltwater effects are available (Dharan 1975).

Moisture diffusion relations are fairly well developed for composites, yet there is rather limited data and limited capability of predicting the effects that the moisture has on fatigue, and particularly on advanced composite fatigue. Various trends have been reported for Gr/E (Kunz and Beaumont 1975, Sumsion and Williams 1975) and for a Gr/G1/E hybrid (Hofer, et al. 1977). A great deal of more recent fatigue research includes service-related moisture testing, particularly in the presence of flaws and other excitations (examples include English 1977-1978, Altman and Olsen 1977, Konishi and Johnston 1978, Dickerson, et al. 1974, Altman, et al. 1979). Such research is viewed as being highly useful to designers. It is noteworthy that these investigations typically seek to mimic anticipated service excitations and flaws, for laminates useful in designing structure. Other investigators have also sought to mimic service loads (Chang, et al. 1977, Sendeckyj, et al. 1977).

### Impact Effects on Fatigue

The effects of impact on Gr/E fatigue strength have been studied (Walter, et al. 1977) and shown to cause damage measurable in several different ways. Similarly, B/Al fatigue strength is shown to be sensitive to impacts (Carlisle, et al. 1975) and to impactor velocity. Some insight into these effects may be gained by perusing pertinent comments in "Notch Sensitivity" and "Impact Strength" sections of this report.

### Post-Buckling Effect on Fatigue Strength

It has been shown that fatigue cycling of Gr/E in the post-buckling load range, applying  $10^6$  cycles at 67% of panel static strength, does not degrade the composite laminate residual strength capability (Bhattia 1976). Much interest in post-buckling fatigue behavior exists in the aerospace design community, and recent funding addresses this topic.

### Load Effects on Fatigue

Early literature indicates that composites are generally more sensitive to fatigue load strain range than metals (Salkind 1972). Subsequent work has shown that excursions into compressive fatigue loads are generally more damaging than purely tensile fatigue loads (Rosenfeld and Huang 1978, Walter, et al. 1977, Sendekyj, et al. 1977, Ryder and Walker 1977, Ramani and Williams 1977). Various discussions and efforts towards identifying key load-related factors which may affect fatigue damage are available. These include studies of mean stress, load range, and maximum compressive stress or stress ratio (Ryder and Walker 1977, Walter, et al. 1977, Sendekyj, et al. 1977, Yang 1978, Roderick and Whitcomb 1977). Again general conclusions are rather difficult to draw in light of the many variables involved (Lauritis 1977).

The behavior of B/A1 is reported to be nonlinear (Chamis and Sullivan 1975) or in other words load history dependent.

By and large, most fatigue data is TT, with TC, and CC following, although some data on torsion (Sumsion and Williams 1975) and shear fatigue (Sims and Brogdon 1977) are reported.

Biaxial fatigue of Gr/E has been reported (Francis 1976), also in the presence of a notch (Francis, et al. 1977).

### Proof Loads and Fatigue Effects

Interest in periodic proof loading as a means of reliability assurance of NDE is evident in the literature, although some practical problems in implementing the method have arisen.\* The idea of applying static loads once in a while to screen out damaged structure which might otherwise fail due to service fatigue loads is well documented (Yang 1977, Yang and Liu 1977, Yang 1978, Yang and Jones 1978, Chou and Croman 1978). Further efforts include research on the effects of the proof loading itself, and studies of the reliability of the method (Yang 1976, Awerbuch and Hahn 1977, Porter 1977, Heller, et al. 1976).

### Improvements in Fatigue Strength

Improvements of B/E fatigue strength by heat treatment under load have been demonstrated (Sun and Roderick 1977) and explained based upon residual stresses. A qualitative discussion of possible fatigue strength improvements in unidirectional metal matrix composites is available (Dvorak and Tarn 1975). Improved fatigue strength of G1/E in water has been demonstrated by using epoxy resins with improved water resistance (Gauchel, et al. 1975).

### Flaw Effects on Fatigue

Recently, there has been increased interest in studies of the effects of flaws on fatigue. Note that our definition of flaws is meant to exclude such things as carefully machined holes, which we prefer to call holes or notches. Some of this work involves flaws which are caused by impact so the reader is referred to "Impact Effects on Fatigue" in this section. Other literature (Konishi and Johnston 1978, Altman and Olsen 1977) is available dealing with man-made flaws, for example, teflon tabs, or dropped tool effects, which are meant to mimic delaminations or other service-induced flaws. Many major aerospace companies presently appear to be doing such testing, and are to some extent publishing their findings.

### LEFM and Fatigue

Various LEFM-derived or LEFM-like modeling efforts for fatigue strength have

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\*Refer to "State-of-the Art Structural Design and Reliability Assurance" section of this report.

have been documented, however, the constraints on applying the models and data to real structure are severe when compared to similar metals technology. The reader is referred to the "Fracture Strength" section of this report, and for convenience pertinent references are listed here (Kunz and Beaumont 1975, Mandell and Meier 1975, Mandell 1975, Kendall 1977, Reifsnider, et al. 1975, Reifsnider, et al. 1977, Chang, et al. 1975). Promising research on delamination growth modeling is underway (personal communication with Dr. Wilkins, Konishi and Johnston 1978).

#### Analytical and Other Fatigue Models

Some attempts at analytical prediction of fatigue effects have been made, but are plagued, among other things, by problems of predicting possible multiple failure modes. Also, attempts at synthesizing analysis to account for multiple modes seem to become rather unwieldy to designers.

A review of fatigue and life prediction attempts is available (Yang and Jones 1978).

One fatigue strength model for predicting unidirectional material fatigue properties is based on measured constituent S-N fatigue curves (Hashin and Rotez 1973).

A stress-based theory extending the Tsai-Hill criteria to fatigue is available (Sims and Brogdon 1977). Another model of notched composites repeats loadsteps on a new specimen description after the results of the previous load have been determined (Kulkarni, et al. 1977). An extension of this work includes a specimen description capable of modeling delaminations generated during loading (Ramkumar 1978), which was previously found to be a deficiency of the model. Some older models may offer useful insight to limited types of fatigue (Tarn, et al. 1975, Dvorak 1975), but the most useful and promising recent efforts generally seem to be empirical models of flaw growth (for example, Konishi and Johnston 1978, personal communication with Dr. Wilkins). One paper (Halpin and Polly 1967) suggests formulation of a general theory of fatigue with visco-

## Design and Manufacturing Variables Effects on Fatigue

Fatigue strength has been shown to depend upon many design and manufacturing variables. A review is available (Salkind 1972) of early research showing dependencies upon fiber volume ratio, orientation, woven vs. nonwoven fibers, fiber modulus, specimen geometry, stacking sequence, and fiber length. In addition other sources are available (for example Hahn and Kim 1976) concerning the following dependencies of fatigue strength upon design and manufacturing variables:

Fiber-related include fiber length (Dharan 1977, Pillar, et al. 1977), fiber endurance limit (Dvorak and Tarn 1975), and woven vs. non-woven fibers (Mandell 1975).

Matrix-related include matrix endurance limit (Dharan 1977, Dvorak and Tarn 1975) and in the presence of water, for G1/E, matrix water resistance (Gauchel 1975).

Laminate-related include fiber orientation (Ramani and Williams 1977, Roderick and Whitcomb 1977), lamina stacking sequence (Hofer, et al. 1977, Ramani and Williams 1977, Sumsion and Williams 1975), specimen size, width, or edge (Sendeckyj and Stalnaker 1977, Ramani and Williams 1977), heat and stress history of B/E (Sun and Roderick 1977), and interface strength (Dharan 1977, Kunz and Beaumont 1975, Hancock and Shaw 1974).

Geometric (machined) include specimen size, width, or edge (Sendeckyj and Stalnaker 1977, Ramani and Williams 1977), and notches, holes, or cutouts. Fatigue loads have in many cases been shown to reduce the notch sensitivity measured as residual strength, of composites, or, in other words, composites are relatively notch insensitive to fatigue loading. The reader is referred to the "Notch Sensitivity" section of this report, however, for convenience, a list of pertinent references follows (Ramani and Williams 1977, Chang, et al. 1977, Roderick and Whitcomb 1977, Reifsnider, et al. 1975, Salkind 1972, Sendekyj, et al., 1977, Papirno 1976). Many experimentalists notice coupon grip problems. Collectively it has been demonstrated that many of these effects may lead to peculiar failure modes in any given experiment. Therefore it is critical to develop testing methods which replicate service structure and conditions.

## Fracture Strength

A moderately large amount of literature is available on fracture strength and fracture mechanics of composite materials. It is assumed that the reader is familiar with concepts fundamental to homogeneous, isotropic, linear, elastic fracture mechanics (Inglis 1913, Griffith 1921, Griffith 1924, and Irwin 1948). This background is also available in more recent summary reviews (Wu 1968, Hardrath 1974, Swedlow 1972, Jones 1975). A review of fracture mechanics efforts in fibrous composites is also available (Broutman and Gaggar 1975).

Throughout the literature concerning composite material fracture mechanics, certain issues appear to have repeatedly received attention. We shall discuss the literature in the context of what seem to be these generic issues:

1. Applicability - Is LEFM applicable to composites and if so, to what extent?
2. Utility - Is LEFM an expedient and useful method applicable to real composite structure?
3. Growth Directionality - Do cracks in composites exhibit self-similar growth? In other words, which way(s) will the crack(s) grow?
4. Notch Size Effect - What causes the notch or hole size effect?
5. Cyclic Loading - Is LEFM useful in considering the effects of cyclic loading?
6. Micro/Macro - Can micromechanical considerations account for macromechanical behavior?
7. Stress Analysis - What is the stress distribution surrounding the flaw or notch?
8. Improvements - How can tougher composites be made?

It is evident that these issues are interdependent, and that more could be listed and restated. For purposes of continuity of discussion, some literature pertinent to notch sensitivity, joint strength, tensile strength with discontinuities, and fatigue strength will also be discussed within this context. The list of issues is not meant to imply a list of research priorities; instead, we seek to provide a sense of what questions research has addressed. In fact, a reader may notice that some research is of limited use to designers and is consequently somewhat academic in nature (issue 2, utility).

First, it is noteworthy that several critical reviews of the application of LEFM to composites are available (Kanninen, et al. 1977, Brinson and Yeow 1977, Zweben 1973). In addition, one typically finds critical commentary in the leading paragraphs of many papers.

Efforts addressing issue (1), applicability, and issue (7), stress analysis, usually reference the derivation of the elastic stress distribution around a through "crack" in an anisotropic, homogeneous plate (Wu 1968). Similar stress analysis is also available in the literature (Sih, et al. 1965). Stress distributions near the "crack" are shown (Wu 1968) to be characterized by:

(1)  $\sigma^\infty/\sqrt{a}$  where  $\sigma^\infty$  = far field stress

$a$  = crack characteristic length

(2) "crack" orientation relative to principal material axis

(3) anisotropic elastic material constants

Additionally, it was predicted that if one limits "crack" configuration to only a "crack" which starts and remains (self-similar growth) aligned with a principal orthotropic material direction, then  $\sigma^\infty/\sqrt{a}$  and  $\tau^\infty/\sqrt{a}$  are sufficient to describe the intensity of the significant stresses around the "crack" tip. For non-aligned "cracks," the deformation mode was shown to have both symmetric and skew-symmetric stress components. Early experiments (Wu 1968) on unidirectional G1/E specimens having center "cracks" with fiber direction showed:

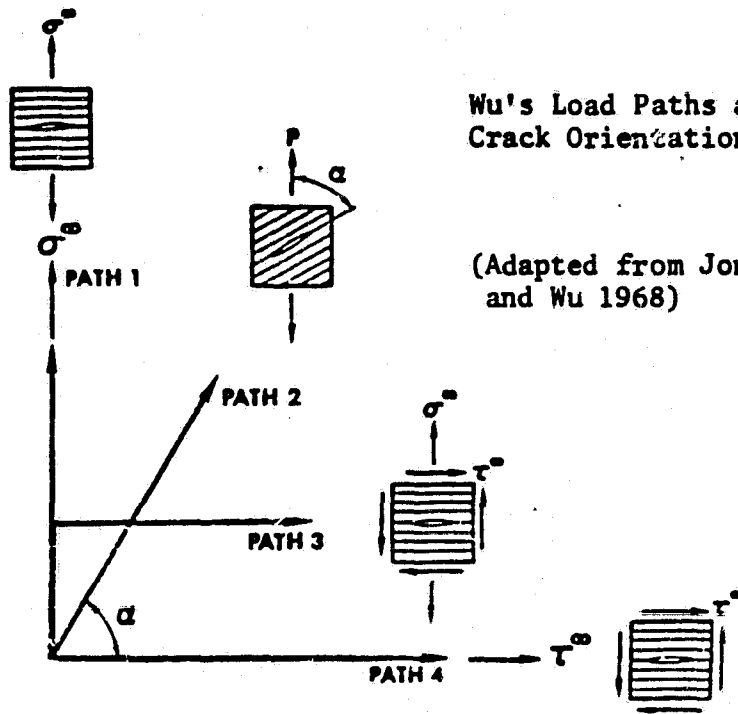
1. self-similar "cracks" are propagated; even with different loading directions (see figure next page),

2. the modes of propagation were open for symmetric loads and sliding for skew-symmetric loads, and
3.  $K_{Ic}$  and  $K_{IIc}$  are indeed material constants, even under combined load conditions.\*

Wu's Load Paths and Crack Orientations

(Adapted from Jones 1975 and Wu 1968)

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Additional tests on unidirectional composite specimens have confirmed GI/E behavior (Sanford and Stonesifer 1971) and have shown to some extent similar trends in behavior for other common composite materials, including B/Al (Peters 1978, Hancock and Swanson 1972, Adsit and Waszczak 1975), Gr/E (Kaelble, et al. 1975, Konish, et al. 1972), C/E (Wright and Iannuzzi 1973), and B/E (Ueng, et al. 1977, Peters 1978\*). Self-similar growth was not evident in one instance (Konish et al. 1972) when the "crack" was initially misaligned with principal orthotropic material directions. The "crack" turned into alignment with orthotropic (fiber) directions.

Throughout the fracture mechanics literature, many specimen configurations appear to have derived from metals technology, including compact tension, single edge notch, double edge notch, and center notch specimens. Examples of each of these are found throughout the cited fracture strength literature and in a few additional papers (Guess and Hoover, 1973, Owen and Bishop 1973).

Although LEFM was demonstrated as being applicable to composite materials, its proven applicability was limited to unidirectional (orthotropic) configurations.

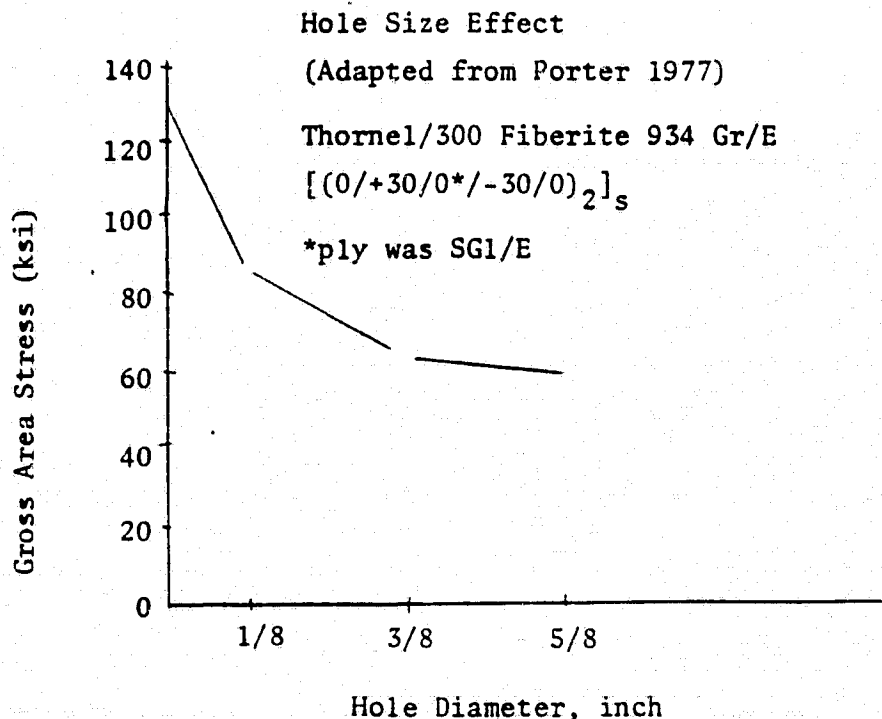
\*See Appendix C.



Since many real composite structures would be expected to consist of angle-ply laminates, one would hope to find some treatments of issue (2), utility, in the literature

And in fact, such literature is evident. However, it would appear that much effort at the time derived from similar metals technology, and as such, issue (2) utility, had not yet been fully considered. In any event, efforts considering angle-ply laminates in addition to unidirectional laminates are available in the literature. In many instances, these machined "through-cracks" are not clearly representative of cracks in service.

Research on angle-ply laminates demonstrates that LEFM can in many instances be used to predict material behavior (Konish, et al. 1972, Konish, et al. 1973, Cruse 1973). Experimental evidence also shows that there is a through thickness cutout, hole, or notch size effect, issue, notch sensitivity in certain specimens (Porter 1977, Walter, et al. 1977, Nuismer and Whitney 1975, Waddoups, et al. 1971, Konish and Cruse 1975, Mar 1976, Hoover and Alfred 1974, Phillips 1974, Zimmer 1972, Holdsworth, et al. 1974, Adsit and Waszczak 1975). Below is typical evidence demonstrating the "hole size effect." Note that larger holes cause greater strength reductions than smaller holes, and that a limiting value in strength reduction is reached as the hole size increases.



Several approaches to explaining the hole size effect are found in the literature; one of which uses LEFM.

The approach using LEFM (Waddoups, et al. 1971)\* begins by noting that the classical elasticity stress concentration factor approach could not explain the effect. It then models the hole as having "intense energy regions" located symmetrically on either side of the hole. The stress field surrounding the hole and its associated intense energy regions was then modeled mathematically as a hole with equivalent cracks. Previously established stress analysis (Bowie 1955) was used. Comments on a physical basis for using the Bowie solution are available (Cruse 1973). It was then shown that the model could be used reasonably accurately to predict tensile strength reductions due to varying hole sizes, given  $K_Q$  and a characteristic length, from test data.  $G$  is assumed to be constant as  $K_I$  changes with  $r$ . The authors anticipated that  $G$  would be a function of laminate orientation and configuration. The authors make comments upon issue (2), utility, as well as (4), notch size effect.

Another approach to explaining the hole size effect does not use LEFM but instead relies upon elasticity and intuitive considerations (Nuismer and Whitney 1975, Whitney and Nuismer 1974). The essence of this approach is that the stress perturbation near a small hole is more concentrated than it is near a large hole, and by intuition stresses therefore have better opportunity to redistribute for a smaller hole. The theory works reasonably well for holes in  $G/E$  and "cracks" in  $G/E$  in various angle ply laminates, and has been extended using both point stress and average stress failure criteria (Nuismer and Whitney 1975, Whitney and Nuismer 1974).

Yet another approach is available which proposes a new material property (Pipes, et al. 197x) and is reviewed in the "Notch Sensitivity" section of this report.

Issue (1), applicability, has been addressed in terms of many subtopics. For  $G/E$  of certain orthotropic angle ply configurations,  $K_{Ic}$  was shown to depend

\*The reader is cautioned that p. 449 of Waddoups, et al. 1971 contains several typographical errors in the equations.

on crack length and to be independent of specimen configuration and thickness (Konish and Cruse 1975). Issue (1), applicability, is often considered to be highly dependent upon issue (3), growth directionality. As a result, many efforts attempting to extend LEFM to non-self-similar growth are found in the literature. For example, one review (Sih and Chen 1973) references previous work on the proposal that

$$G_c = G_I + G_{II}$$

where  $G_c = G$  composite

$G_I =$  mode I energy

$G_{II} =$  mode II energy

A similar expression is proposed for polymeric solids (Irwin 1968). Further efforts (Sih and Chen 1973) propose that

- (1) crack initiation occurs in a calculable direction
- (2) crack extension occurs when strain energy density reaches a critical value.

Kanninen, et al. 1977 review more non-self-similar growth studies and failure theories. Additional research treating mixed-mode fracture in Gr/E is available (Morris and Hahn 1977\*, McKinney 1972). Combined torsion and tension loading has been treated (Saint-John and Street 1974).

Issue (5), cyclic loading, might also be considered a subtopic of issue (1), applicability. The utility of LEFM in predicting real world structural composite behavior under cyclical loading is rather limited, and it may be useful for some researchers to reconsider issue (2), utility, in light of the results available on issue (7), cyclic loading. [Some insight on issue (6), micro/macro is offered in the literature (Kaelble, et al. 1975)]. Such consideration has apparently led to promising work on a simple delamination growth model (private communication with Dr. Wilkins). An energy release rate approach to edge delamination is reported in the literature (Rybicki, et al. 1977).

LEFM combined with micromechanical considerations of fiber S-N curve behavior have been used to model tensile fatigue behavior of Gl/E and Gl/PE (Mandell 1975, Mandell and Meier 1975). Data on Gr/E compressive fatigue behavior is fit to  $\frac{da}{dN} = B(\Delta K)^n$  (Kunz and Beaumont 1975) and shown to give values of  $n$  from 4 to 26 depending on fiber orientation, layup, etc. A model predicting residual strength of notched B/Al and B/E after O-T fatigue of varying frequencies is proposed (Reifsnider, et al. 1975) which may be useful in considering strength increases which sometimes occur after cyclic loading. Behavior of B/Al under cyclic loading is modeled (Kendall 1977) via LEFM in conjunction with a local heterogeneous region (LHR). This LHR concept is also discussed elsewhere in the literature (Kaelble, et al. 1975). Other fracture mechanics treatments of cyclic loads are found for Gr/E (Papirno 1977) and Al/E (Thornton 1972). LEFM has proven to be useful in predicting impact damage (Husman, et al. 1975).

A number of efforts have addressed issue (6), micro/macro. Many efforts appear to have been made attempting to quantitatively predict macromechanical fracture behavior based upon micromechanical events, whereas others appear to be aimed more at gaining qualitative insight. Debonding of fiber ends has been suggested as an energy absorption mechanism, following micromechanical stress analysis (Sendeckyj 1974). Fracture of B/Al with holes and slits has been approached using micromechanics (Mar and Lin 1977). Energy of work or fiber pullout and debonding are modeled (Kelly 1970). Other micromechanical papers on interface effects are referenced by Kanninen, et al. 1977. One paper (Olster and Jones 1972) proposes that the toughness of Gl/E is due to the tensile energy stored in the debonded portion of fibers near the fracture surface. Kanninen, et al. 1977 reference a study of B/E toughness. Rule of mixtures attempts (Olster and Jones 1972) at fracture stress are also presented in the literature. Some correlation between fiber volume fraction and failure modes has been discussed (Sih, et al. 1973). A homogeneous model having a local heterogeneous

region near the crack tip is discussed (Kanninen, et al. 1977a and b). Visco-elastic effects have been considered (Schapery 1976, Brockway and Schapery 1978). Of some relevance, particularly to resins and frequency effects are failure theories in amorphous polymers (Valanis and Yilmazer 1978).

Somewhat in regard to issue (1), applicability, and issue (2), utility, correlation of Charpy energy with  $G$  has been used in researching B/A1 turbine blade FOD behavior (Carlisle, et al. 1975). The instrumented Charpy test has been used to obtain dynamic fracture toughness measurements (Hoover and Guess 1973). Other investigators have also worked with Charpy data (Slepetz and Carlson 1975, Beaumont and Sorver 1975), not necessarily in a LEFM context.

Further efforts can be considered applicable to issue (2), utility, including a method of estimating orthotropic angle-ply laminate  $K_Q$  from lamina and constituent properties (Konish, et al. 1973). Also, efforts to obtain properties by the resistance method (Morris and Hahn 1977, Gaggar and Broutman 1975) and K-calibration technique (Sun and Prewo 1977, Awerbuch and Hahn 1978) are available.

The complexities of elastic stress analysis of heterogeneous material having multiple orthotropic lamina stacked up at different orientation angles with inherent flaws are formidable. More rigorous analyses based on less simplifying assumptions have been made since early efforts (Wu 1968). Still, a great many assumptions are required. Stress analysis efforts include 3D solutions to a through crack (Hilton and Sih 1975, Wang, et al. 1975a, Wang, et al. 1975b, Atluri, et al. 1975) and also a part-through crack (Ko 1978). Analysis is available for a crack in an internal lamina (Arin 1978). Stress intensity is shown to be coupled with specimen configuration, and means of minimizing the effects are offered (Konish 1975). Heterogeneous material crack tip analysis is available (Nuismer and Sendeckyj 1977). Experimental determination of stress intensity factors by evaluating the  $J$  integral via strain measurements has been

reported (Tirosh and Berg 1974). An analytical model of elastoplastic crack propagation in transversely loaded unidirectional B/A1 is available (Adams 1974\*).

Consideration of hygrothermal moisture effects on unidirectional Gr/E toughness (Kaelble, et al. 1975) gives some insight into the effects of other excitations, part of issue (1), applicability, on fracture strength. The toughness is shown to irreversibly increase 2 to 5 times after severe hygrothermal exposure. A micromechanical interface degradation model, issue (6), micro/macro, is proposed as an explanation. Some effects of saline water on Gr/E compressive fatigue have been studied (Kunz and Beaumont 1975). Some early comments (Irwin 1968) on suspected moisture effects and crack growth in polymeric solids as compared to metals are available.

Studies of dependencies of fracture toughness upon design and manufacturing variables (Hoover 1976, Mandell, et al. 1974, Mandell, et al. 1975, Prewo 1978, Ellis and Harris 1973, Wabach, et al. 1968) have addressed issue (1), applicability, and may in some cases be useful in developing improved laminates, issue (8), improvements. B/A1 fracture mechanics data is available for several categories of laminates (Underwood 1974).

Efforts aimed towards improving laminate toughness, issue (8), via crack arrestment are available (Chu, et al. 1976, McKinney 1972\*, Bhattia and Verette 1975, personal communication with Poe). An excellent summary and review (Sendeckyj 1975) of such efforts and methodologies is available, along with development of LEFM crack arrestment design criteria. It is recommended that notch size be considered.

It would appear that, for purposes of composite material structural design, the fracture mechanics approach has proven to be more limited in applicability when compared to similar metals technology. However, despite much controversy concerning aforementioned issues (1-8), useful and expedient methods of designing composite bolt holes, small (i.e., less than 1" for Gr/E) cut-outs, and notches

have been derived from LEFM technology. One therefore finds some discussion in the literature of composite material fracture mechanics in the context of reliability and design philosophy (Halpin, et al. 1973, McCarthy and Orringer 1975). Some interesting general comments on reliability are also available (Bondi 1978, Maxwell, et al. 1975).

The reasons LEFM methods are more limited in application to composites are:

- (1) LEFM specimen through cracks or notches are seldom representative of flaws or excitations found in service (with the exception of small holes and cutouts).
- (2) self-similar crack extension necessary for LEFM methods doesn't always occur, and
- (3) heterogeneous materials have multiple possible failure modes, with several modes occurring simultaneously.

The literature often appears to provide appealing and self-consistent results, but the bulk of it is confined to a narrow range of problems relative to those occurring in service. It would be useful to expand the utility of the problem types dealt with in the course of traditional research. One mechanism by which the scope of problems dealt with could become more utilitarian is for funding agencies to consistently set research priorities in terms of service experience as opposed to prior research.

### Impact Strength

There is a great deal of aerospace interest and literature (witness ASTM STP 568 "Foreign Object Impact Damage to Composites") concerning the impact strength of composite materials, for several key reasons.

Contact with foreign objects (one often sees "FOD" for foreign object damage in the literature), for example birds (soft bodies) and hail, rocks, and dropped tools (hard bodies) is normally anticipated in service, and experience has shown that some tough problems exist. Composite turbine rotor blades have proven to be susceptible to bird ingestion (see "Engines" listing in "Aerospace Applications" section of this report). Impacts due to hail, runway debris, pebbles, bird strikes, and dropped tools can in some cases cause laminated (and, for fairness, also metal alloy) structural damage. Thus, much interest has centered around improving and understanding composite impact strength.

Despite such efforts, the soft body impact rotating blade problem persists. Also, efficient means of assessing suspected damage on composite skins after excitation, such as flight through a severe hail storm, are in their infancy.

Much of the pertinent literature is a combination of experiment and analysis. A great deal of research concerning impact strength as it relates to turbine blades is done by General Electric, NASA Lewis Research Center, Pratt and Whitney, and USAF Wright-Patterson AFB. For example, activities on B/A1 blades and specimens (Salemm and Yokel 1978, Oller, et al. 1977, Carlisle, et al. 1975), metal composites (Winsa and Petrusek 1973), and many other composites including Gr/E (Chamis, et al. 1972, Preston and Cook 1975, Sun and Sierakowski 1975) are found in the literature. These papers usually include studies of hard and soft body impacts representative of service conditions.



Part of the impact strength literature is more oriented towards the composite laminated skin type of application as might typically be found on lift or control surface structure. Such literature (Miner, et al. 1975, Greszczuk and Chao 1977, McQuillen and Gauss 1976, Rhodes 1974, Broutman and Mallick 1974, Oplinger and Slepetz 1975) typically reports experiments in which plates or representative structures are impacted with balls or objects representative of rocks, hail, birds, tools, or whatever. In addition, literature (Avery and Porter 1975, Soares and Whiteside 1975, Olster and Roy 1974) is available comparing the effects of small arms ballistic impact on Gr/E and B/E composites with those effects on traditional metal alloys, for purposes of military aircraft design. Reference to previous pertinent work is also available (Cristescu, et al. 1975). Residual tensile strength after penetration by a bullet has been shown to correlate quite well with a model which represents the hole as a drilled hole of the same size (Reference [2] in Husman, et al. 1975) in B/E, Gr/E, and Gl/E. Models for predicting residual strength after impact at less than penetration velocity are available (Husman, et al. 1975, Averbach and Hahn 1976) which rely upon energy and LEFM considerations. An equivalent hole or "flaw" is derived for the impact. Data on tensile strength after impact is found in other papers (Lifshitz 1976).

Development of improved impact strength hybrid laminates for helicopter drive shafts is reported in the literature (Figge, et al. 1974).

Experiments have shown that impact strength or energy depends upon many factors, including matrix and fiber elongation to fracture, matrix modulus, fabrication process, fiber and void volume ratios, residual stresses (Charmis, et al. 1972), fiber orientation angle, stacking sequence, specimen geometry (Broutman and Mallick 1974a), hybrid fiber type and interlaminar shear strength (Broutman and Mallick 1974b), strain energy to failure of the reinforcement (Husman, et al. 1975), impactor velocity (Broutman and Rotem 1972), fiber strength, fiber modulus, fiber orientation, and stacking sequence (Gresszczuk and Chao 1977), tensile stress-strain characteristics of the fiber (Novak and DeCrescente 1972),

and fiber-matrix interface and matrix shear strength (Broutman 1976, Carlisle, et al. 1975) and temperature (Sharma 1978) to name a few. Various arguments concerning the dominance of a given effect are available, however, one must consider each argument in the unique context of the particular specimens and experiment which may have been run. It can perhaps be anticipated that such dominance would vary somewhat depending upon the specifics of each specimen, material, and experiment.

In many experiments, impact strength or energy appears to be measured in various manners derived from metals technology, including Izod impact testing (Winsa and Petrasek 1973, Chamis, et al. 1972) and some forms of Charpy testing (Adams 1977, Prewo 1972, Broutman 1976, Novak and DeCrescente 1972, Beaumont and Server 1975, Carlisle, et al. 1975, Helfenstine 1977, Winsa and Petrasek 1973). In any event, it is evident that multitudes of diverse impact specimens, experimental methods, and failure modes have been and can be generated. The utility of some of these experiments is not clearly evident, when compared to others which are of obvious interest to structural designers.

Experiments and some models which comparatively rank different composite materials in terms of impact strength (Beaumont, et al. 1975, Chamis, et al. 1972, Novak and DeCrescente 1972, Greszczuk 1975) are available. In general, G1/E and K/E, sometimes in hybrid form, are shown to be more impact resistant than Gr/E. The relatively low strain to fiber failure of Gr/E is offered as a logical reason. A designer is often interested in a family of material properties, so that more useful rankings might also consider for example impact strength per unit weight or per unit stiffness. SG1/E face sheets on Gr/E are shown (Oplinger and Slepetz 1975) to offer improved impact strength.

Experiments on K49/E (Miner, et al. 1975) have shown that impact strength per ply decreases as the number of plies increases, and that increased interweaving raises impact strength; hence style 281 fabric will have greater impact strength than style 181.

Failure mechanisms in G1/E subject to cylindrical blunt end penetrators have been studied (Cristescu, et al. 1975) along with a review of pertinent previous literature.

Of significant interest to designers may be research (Rhodes 1975, Rhodes, et al. 1976, Rhodes, et al. 1978, Walter, et al. 1977) showing that impacts occurring before or during compression loading of Gr/E structure can severely degrade compressive strength and other properties (Walter, et al. 1977) of the structure.

A multitude of impact models have been proposed. Most attempt to model situations in which a planar surface or beam is impacted normal to its surface (Sun and Sierakowski 1975, for example), although treatment (Mortimer, et al 1975) of in-plane impact measurements and analysis of longitudinal waves was noted. Mortimer references many of the out-of-plane impact research papers.

Most of the out-of-plane impact models are analytically derived, dynamic models, with some empirically derived models also being available. For example, improved impact strength laminates have been derived (Chamis, et al. 1972) using several semi-empirically derived equations. The equations depend upon failure mode and upon micro and macromechanical considerations. Models predicting relative ranking, failure mode sequences and experimental data for G1/E, Gr/E, and B/E are available (Greszczuk 1975).

Most analytical models ignore through the thickness effects and are set up to give some information concerning out-of-plane vibration and wave propagation. One computerized dynamic analysis (Kibo and Nelson 1975) accounts for some through the thickness effects. The model appears to be capable of giving some information on wave propagation along the laminate plane as well as through its thickness. Thus, low frequency, long wavelength as well as higher frequency, shorter wavelengths may be accommodated by this model. Ply variations may also be modeled.

A test case of a simple B/E laminate subject to ramp loading impact was run and showed the transient passage of potentially damaging stress waves. The authors suggest need for experimental work and comparisons to confirm the model's utility as a design tool. Data on B/E is available (DeRosset 1975, Greszczuk 1975).

Several analytical models of impact are available based upon conservation of energy. One (Sun 1977) equates kinetic energy of a rigid ball before impact with kinetic energy of vibration and energy of damage after impact. An unloading path is assumed, and the energy going into damage is calculated using force from the Hertzian contact law and indentation. The expression for vibration energy is not analytical. The model is limited to hard body impact and is fit to data on G1/E. The author offers some intuitive comments concerning soft body impact.

Another analysis (Greszczuk and Chao 1977) assumes that the impact pressure loading is a function of time and the stress a function of that pressure. Damage is then derived from the stresses by the distortion energy theory. The authors experiment with Gr/E impacted by steel balls and find that damage resistance varies with several parameters.

Maximum strain was found to be proportional to velocity and to square root of impactor mass in one analysis (Chou 1976). The same author also outlines several other approaches to understanding impact behavior.

A comparison of three analytical force models, Hertz, impact-momentum, and modified Hertz is available (Preston and Cook 1975), along with experiments of Gr/E impacted by gelatin, ice, and steel. Stress waves are considered.

It appears that some major differences exist between corporations in the levels of pursuit of composite bladed engines. Considerable controversy, some of which is based upon some previously cited technical research, arises in attempting to explain such differences.

## Joint Strength

Composite material joints and hence joint literature may be generally classified as bonded, mechanically fastened or bonded and mechanically fastened. Some excellent reviews and references are available (Advanced Composites Design Guide, Third Edition, 1973 Vol. I Sections 1.3 and 1.6.3 and Vol. II Section 2.4, Cruse, et al. 1972, Eisenmann 1976, Jones 1975). Additionally, it is reported (notes from 1,2 November 1978 Bergamo Center Meeting) that Gustavson of Ft. Eustis has completed a literature search on composite material joint design.

Due to the notch sensitivity of many advanced composites (see "Notch Sensitivity" and "Fracture Strength" sections of this report), use of mechanical fasteners and their required holes imposes quite a penalty (up to 40%) on strain allowables in many aerospace applications. Despite this impetus to avoid holes and hence mechanical fasteners, the reliability of bonded joints does not appear to be as clearcut as that of fastened joints, and hence many companies are using fastened or fastened and bonded joints (Waddoups 1977). In any event, there appears to be much difference in joint philosophy in the industry. It appears that much of the controversy may stem from the use of deterministic vs. probabilistic models and analysis.

A sense of the interest and activities towards improving bonded joint strength is available along with a state-of-the-art review of the physical understanding of bonding (Drzal 1977). Hygrothermal effects on bonded joint strength have been studied both experimentally and analytically (Vinson and Pipes 1976, Weitsman 1977). A reliability model (reference [1] in Berens and West 1975) and other literature pertinent to bonded joints is reviewed and shown to be useful in predicting the effect of accelerated fatigue testing at a higher temperature (Berens and West 1975). A test method has been designed (Renton and Vinson 1975) to measure in-situ bonded joint properties in an attempt to get around problems of using neat resin measurements (Brinson, et al. 1975). Fracture mechanics approaches to bonded joint strength are found in the literature. A

single-edge-notch type of specimen has been used (Trantina 1972a). Combined modes I and II crack extension have been treated (Trantina 1972b). Treatment of fracture mechanics and time dependent joint strength is available (Knauss 1971). Stress analysis of a stepped scarf-type joint is available (Erdogan and Ratawani 1971). Some of the trends exhibited in aerospace bonded joints are also evident in bonded joints intended for housing construction (Masters and Reichard 1975).

A straightforward design method for mechanically fastened joints is clearly documented (Eisenmann 1976). It predicts failure load and failure location in single or multiple fastener joints, and accounts for geometry, fastener diameter, laminate mechanical properties, and complex load applications. The methodology derives from LEFM.

Early single fastener (Waszczak and Cruse 1971) and multi-fastener joint (Cruse and Swedlow 1971) design philosophies determine bolt load distribution among the various fasteners in the joint, calculate or estimate stresses at the perimeter of the fastener hole, and compare the stresses to material allowables using appropriate failure criterion. Multiple failure modes are considered, and ply strains and stresses are computed and compared to criteria.

Much of the bolted joint literature is directly concerned with development of bolted joints for hardware design purposes, hence such aircraft as the F-111 (Ashton, et al. 1972) and DC-10 (Elliot 1977) are mentioned. Analysis and test of a (Reed and Eisenmann 1975) Gr/E scarf joint simulating flight service spectrum tension and compression fatigue loading showed the usefulness of a wearout model (reference [1] in Reed and Eisenmann 1975) in predicting lifetimes and residual strength. The effects of moisture and temperature on joints (Wilkins 1977, Elliot 1977, personal communication with M. S. Rosenfeld) have received increasing attention recently. Hygrothermal excitation is shown (Wilkins 1977) to degrade Gr/E bolted joint strength up to about 18%.

An excellent review of many early studies addressing improvements in mechanical fastening methods is available (Padawer 1972), along with demonstrations

of significant improvements in Gr/E joint strength made possible by colaminating boron films. Some improvements in joint strength have been demonstrated by using interference fit fasteners and/or clamp up forces (personal communications with the industry), however, there is much debate concerning the economics of the improvement, and backside delaminations are sometimes a problem. There is lots of activity on composite fasteners (personal communication with Daschal), including a March 1978 meeting, a materials standard for NAS type 65 fasteners, and an AIAA report.

One uncommon paper (Gillespie and Pipes 1978) reports finite element stress analysis and experimental verification studies of integrally joined spar panel stiffeners. Studies are made on Gr/E with and without T-shaped titanium inserts. Similar work has been reported at NASA Langley.

C-2

### Notch Sensitivity

Many papers dealing with notch sensitivity are available in the literature. The composites treated, including Gr/E (Kocher and Cross 1972, Cruse 1973, Olster and Roy 1974, Waddoups, et al. 1971, Bhattia and Verette 1975, Brinson and Yeow 1977, Mar 1976, Whitney 1976, Cruse, et al. 1972, Daniel 1976, Greszczuk 1972, Yang 1976, Nuismer and Whitney 1975, Whitney and Kim 1976 and 1977, Kibler 1976, Ramani and Williams 1977, Walter, et al. 1977, Chang, et al. 1977, Sendekyj, et al. 1977, Kunz and Beaumont 1975, Porter 1977), B/E (Bhattia and Verette 1975, Rowland, et al. 1974, Olster and Roy 1974, Sun and Roderick 1977, Cruse, et al. 1972, Greszczuk 1972, Roderick and Whitcomb 1977, Kulkarni, et al. 1977, Reifsnider, et al. 1975, Reifsnider, et al. 1977), B/Al (Mar 1976, Adsit and Waszczak 1975, Reifsnider, et al. 1975, Reifsnider, et al. 1977; Prewo 1974), G1/E (Greszczuk 1972, Nuismer and Whitney 1975, Porter 1977), and B/Titanium (Prewo 1974) are generally shown to be notch sensitive to monotonic loads and relatively notch insensitive to cyclic loads. Most loads treated are uniaxial tensile loads, however, there is some treatment of biaxial (Daniel 1976), shear (Greszczuk 1972), and compressive loads (Sendekyj, et al. 1977). Both unidirectional and angle ply laminates are treated.

Notches of many forms, shapes and sizes have been treated. Most notches treated are specimen-centered round through-holes or narrow through-slots (sometimes called cracks), although treatments of some other notch geometries including angled slits (Whitney 1976), surface notches (Sendekyj, et al. 1977), single edge notch (Olster and Roy 1974), and double edge notches (Brinson and Yeow 1977) are available. The effects of smooth and rough round, elliptical, and square holes in B/E on tensile strength have been reported (Rowlands, et al. 1974). Plate width and thickness effects are also reported by Rowlands. Treatment of countersunk holes, slits, and angle slits is available (Nuismer and Labor 1978). Notches are described by a variety of names; in addition to those indicated above, the term "flaws" is sometimes used (Reifsnider, et al. 1977, Porter 1977). This wide



range of terminology in use is at least confusing but, more importantly, a reader could be misled into thinking that they are treating a problem different from the one actually at hand. Standardized descriptions are needed.

Two popular approaches to explaining composite notch sensitivity are evident: (1) fracture mechanics and (2) elasticity. Early analyses are reviewed in one paper (Beaumont and Phillips 1972). Discussion of these two approaches in relation to explaining monotonic tensile load notch sensitivity can be found in the "Fracture Strength" section of this report. A review of cyclic loading notch sensitivity literature follows.

A phenomenon observed in some but not all notched composite tests is an improvement in residual (usually tensile) strength after cyclic loading. It is important for the uninitiated reader to note that this particular and other related phenomenon have received treatment in the literature under many different guises, including notch sensitivity, residual (notched) strength, and fatigue strength. Consequently, this phenomenon could also presently be discussed within several different contexts and under several different headings, such as (residual) tensile strength of notched composites, or notch sensitivity, or fatigue strength of notched composites. The choice of heading to some extent depends upon the quantifiable property of interest, i.e., tensile strength, stiffness, compressive strength, number of cycles, or whatever. We choose to discuss this body of literature in the context of "notch sensitivity," and suggest that if need be, the reader may be able to define (an equation for) notch sensitivity which best suits his own interests. Examples of quantified notch sensitivity are readily available for traditional metal alloys (Shigley 1972).

Efforts to correlate notch effects for cyclic and in many cases monotonic loading using LEFM or LEFM-derived methods are available (Kunz and Beaumont 1975, Porter, 1977, Reifsnider, et al. 1975, Cruse, et al. 1972, Adsit and Waszczak 1975), and generally show reasonably good agreement. LEFM may not be too satisfying in accounting for the frequency effect (Reifsnider, et al. 1977, Reifsnider, et al. 1975).

Analytical elasticity, plasticity, and mechanics of materials approaches to account for notch sensitivity under cyclic loading are available (Whitney and Nuismer 1974, Kulkarni, et al. 1977, Ramkumar, et al. 1978, Sendeckyj, et al. 1977, Rybicki and Hopper 1974). One extension (Rankumar, et al. 1978) of this work includes delamination effects. In general, these types of cyclic loading notch sensitivity models appear to be of rather limited applicability in design and reliability assurance situations when compared to similar metal alloy technologies. The strain energy density theory has been applied to composites with slits (Sih, et al. 1975). Stress concentration studies have been reported (Gerstner and Dundurs 1969, Kulkarni, et al. 1973, Chou, et al. 1977). Stress analysis near a hole is available (Konish and Whitney 1975).

Already, improved material properties and designs have been developed using the results of such research. For example, crack arrestment concepts (Bhattia and Verette 1975, Sendeckyj 1975) have been developed based upon LEFM, improved B/E notched fatigue strength by heat treatment under load has proceeded semi-empirically (Sun and Roderick 1977) and reinforced holes and cutouts in Gr/E have been developed (Kocher and Cross 1972) based upon elasticity. Notch sensitivity of a composite containing a Boron film reinforcing layer has been reported (Padawer 1974).

Miscellaneous effects upon notch sensitivity such as stacking sequence (Rybicki and Schmueser 1978, Whitney and Kim 1977), cyclic load range (Sendeckyj, et al. 1977), temperature (Sendeckyj, et al. 1977, Whitney and Kim 1976), and saltwater (Kunz and Beaumont 1975) have been addressed.

Analysis of an orthotropic laminate containing a layer with a crack is available (Arin 1977).

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Shear Strength

Shear strength is the focus of a moderate number of papers. Most of these deal with interlaminar shear strength, while a few treat in-plane panel shear strength and transverse shear strength. These few exceptions tend to be hardware-design oriented papers. There is additional literature concerning bonded joint shear strength (Vinson and Pipes 1976, Masters and Reichard 1975) which is more appropriately grouped with joint strength.

Of the hardware-oriented papers, one (Wesselski 1976) reports transverse shear stress failure of overwrapped K49 composite applied to pressure vessels; yet the shear failure mode has not clearly caused structural failure. Other hardware-oriented papers include in-plane and interlaminar shear testing of B/Al with temperature excitation for use on a DC-10 (Elliot 1977), and analytical design and experimental testing of two large Gr/E skinned aluminum honeycomb core sandwich shear webs (Bush 1975). In-plane shear and other loading has also been treated analytically. A paper reporting studies of minimum weight shear panels with holes is available (Foye 1970). Stress analysis using the Airy stress function of laminates containing holes is available (Gresczuk 1972). The same author presents strength analysis based upon modified Hencky-Von Mises distortion energy theory. Another paper (Weller 1977c) reviews other in-plane analytical tools and concludes that they're not too accurate.

No one test method has seemed to gain acceptance as the standard shear strength test method. ASTM D2344-67 is reported (Berg, et al. 1972) to underestimate maximum shear stress by about 100%. The authors suggest using the test only for screening purposes, and using torsion tests for precise shear measurements. An apparatus reportedly (Hill and Anderson 1972) capable of applying a pure moment is suggested as being useful in measuring interlaminar shear strength. In-plane shear testing methods for Gr/E and in some cases B/E have also shown some dispersion (Slepetz, et al. 1978, Sims 1973, Bush and Weller 1978, Bush 1975,

Weller 1977c, Weller 1977d). Two test methods for determining the in-plane shear stress-strain response of a unidirectional composite have been reported (Sims and Halpin 1974). These methods are the uniaxial tension test on a  $\pm 45$  degree laminate and the rail shear test on a 0/90 degree laminate. Further discussion of the rail shear test is available (Whitney, et al. 1971). Sims and Halpin also study and report viscoelastic characteristics. Interlaminar shear strengths of unidirectional B/A1 measured by both three point bend and short beam shear methods have been measured and compared (Peters 1978). A review of interlaminar strength tests is found in a recent paper (Harris and Orringer 1978). The three point bend type of specimen has also been used to study C/E interlaminar shear strength under impact loads (Sayers and Harris 1973). Other methods are reviewed in Chapter 2.8.2 of Jones 1975.

Interlaminar shear strength is shown to correlate with compressive strength (Davis 1975). Analysis of interlaminar shear stresses has received treatment in the literature. A pertinent literature review is available in a recent analytical paper (Hsu and Herakovich 1977). A perturbation solution method is used and curves of interlaminar stresses near the free edge are plotted. The solution is offered as an improvement on a previous finite difference solution (Ref. [2] in Hsu and Herakovich 1977). Reviews of this solution, experimental confirmations, and implications are available in Ch. 4.6 of a handy textbook (Jones 1975). Some analysis on panel flutter (Material Sciences Corporation 1976) has indicated damping and other vibration parameters to be functions of interlaminar shear and transverse shear. Limit and other analyses are available for unidirectional composite shear behavior (Shu and Rosen 1967, Hahn 1973, Adams and Doner 1967).

The Tsai-Hill static failure theory has been extended in an attempt to analyse Gr/E and Gl/E interlaminar shear fatigue effects (Sims and Brogdon 1977). Experimental study of Gr/E, Gl/E, and B/E (Pipes 1974) interlaminar shear fatigue has shown up to 50% degradation in shear strength after  $10^6$  cycles.

Shear buckling behavior has received limited treatment in the literature (Whitney 1969, Kaminski and Ashton 1971).

Correlations of shear strength with ultrasonic attenuation (Hayford, et al. 1977) and bulk compressibility (Smith, et al. 1973) have been reported.

Studies of the effects of several excitations on shear strength, including moisture (Kaelble, et al. 1975, Browning and Hartness 1974, Browning, et al. 1977, Broutman 1976, Kaelble and Dynes 1977), temperature (Unknown NBS 1978, Browning and Hartness 1974, Kaelble and Dynes 1977), and time (Foye 1975, Sayers and Harris 1973) are available in the literature. Interlaminar shear strength of unidirectional Gr/E shows degradation of up to 50% after 200 hours of water immersion or humidity exposure (Kaelble, et al. 1975). Other effects are studied, and many changes in material properties are shown to be irreversible. An expression for interlaminar shear strength is offered, and many pertinent references are cited. Interfacial bond degradation is offered as a possible explanation of such effects (Kaelble, et al. 1975, Kaelble and Dynes 1977, Browning, et al. 1977). Matrix swelling and lowering of  $T_g$  are also offered as possible explanations (Browning, et al. 1977). What may be regarded as conflicting trends concerning moisture effects are available (Broutman 1976) for certain materials. It is apparent that great difficulty may arise in attempting to duplicate some experiments reported in the literature, particularly those that are resin-sensitive. Many details concerning prepregs or fibers and resins, processing, and experimental procedure are often omitted from the literature. There is the additional problem of possibly inconsistent prepreg characteristics from batch to batch, in which case the reporting of a given prepreg name and processing may not suffice in characterizing the material.

Cryogenic effects on shear strength for several popular materials are reported (Unknown NBS 1978) as being rather encouraging.

Analytical micromechanical considerations of creep phenomenon have been used to predict shear strain as a function of time (Foye 1975). Water exposure

over a long period of time is also shown to correlate with interlaminar shear strength (Kaelble and Dynes 1977). The authors argue that their epoxy is chemically irreversibly affected after a water soak, which is contrary to the comments some chemists have been made (personal communications) concerning popular epoxy resins. Kaelble and Dynes also report study of many NDE methods as means of monitoring shear strength degradation.

Some design and manufacturing variables have been shown to effect shear strength, including misaligned fibers (Claus 1972) and void content (Sendecky 1976).

## Stiffness

Many papers deal with laminate stiffness, both directly and indirectly. Any paper presenting complete stress-strain data can be used to obtain an effective laminate modulus or stiffness in the given loading mode. Such papers typically do not focus upon stiffness, hence they are said to deal indirectly with laminate stiffness. These papers are not reviewed here, however, a few notes of caution are in order. A reader will find that many papers cannot be used to deduce such information because of incomplete data reporting, lack of plotted results, and lack of effective means of accessing the appropriate literature.

Fortunately, direct treatment of stiffness in a clear and useful manner is available in the literature. This may follow from the criticality of stiffness in many applications, the popularity achieved by composites due to their high specific stiffness (Petrie 1975), and the relative ease with which stiffness can initially be measured and analyzed. Designers generally have reasonably good stiffness design tools available, however, some questions remain including those concerning long-term effects on stiffness. Aerodynamic tailoring design rationale are reported in the literature (Konishi, et al. 1976).

Analysis of stiffness on the lamina level is clearly treated (Jones 1975) in terms of the "Mechanics of Materials Approach to Stiffness," and "Elasticity Approach to Stiffness," along with a "Comparison of Approaches to Stiffness." On the laminate level, the same author reviews classical lamination theory and other pertinent work in Chapter 4. Some pertinent work on plates is found in Chapter 5. Jones' book is suggested as an excellent reference for lamination theory and other topics. Some similar information is available in earlier literature (Ashton, et al. 1969, Advanced Composites Design Guide, Third Edition 1973).

A 3D finite element method of determining laminate effective modulus has been reported (Kang and Rentzepis 1974). Graphical methods have also been reported for determining stiffness and tensile strength (Hahn and Tsai 1974). The effect of voids on tensile and compressive laminate moduli is considered in a simplified elastic formulation (Cohen and Ishai 1967). 3D laminate moduli are treated in an early paper (Enie and Rizzo 1970). Other available papers treating stiffness or moduli (Boucher 1974) are concerned with torsion and tension (Ishikawa, et al. 1977), fiber waviness effects (van Dreumel and Kamp 1977), random filament packing (Adams and Tsai 1969). Short fiber composite effective stiffnesses are treated in the literature (Halpin 1969, Christensen and Waals 1972).

Stiffness as a means of fatigue damage measurement was suggested in the early '70's (Salkind 1972). Many papers concerning fatigue loading measure stiffness to assess damage, and many different excitations, design and manufacturing variables, and loading modes have been treated (Roderick and Whitcomb 1977, Hofer, et al. 1977, Kasen, et al. 1977, Reifsnider, et al. 1975, Sims and Brogdon 1977, Awerbuch and Hahn 1977, Salkind 1972, Heller, et al. 1976, Adams, et al. 1975, Reifsnider, et al. 1977, Chamis and Sullivan 1975, Stinchcomb, et al. 1975, Sumsion and Williams 1975, Dharan 1977, Ritchie and Rosinger 1975). The reader is referred to the "Fatigue Strength" section of this report for commentary on these papers.

Nonlinearities in stress-strain curves of many composites are evident (Jones 1975, Advanced Composites Design Guide, Third Edition 1973), especially in laminates having high matrix shear loading.

Studies of Boron fibers and composite stiffness have indicated existence of nonlinearities (Chamis and Sullivan 1975, 1975\*, Elliot 1977, DiCarlo 1977). Computerized analysis of B/Al tangent modulus including effects of residual matrix stresses is available (Chamis and Sullivan 1975\*), along with anelastic Boron fiber analysis (DiCarlo 1977).



Some attempts at explaining nonlinear effects have been made based upon creep phenomenon (Foye 1975, DiCarlo 1977), and failed ply and other phenomenon (Sendeckyj, et al. 1975, Jones 1976). The knee in a Gr/E tensile stress-strain curve has been shown to correspond to initiation of microcracks in transverse plies (Hennecke, et al. 1976). G/E and C/E matrix cracks have been shown (Adams, et al. 1975) to degrade stiffness and improve damping, and have also been shown to be measureable by vibrational means. An analytical model of modulus of 3D and other array composites is available (Rosen, et al. 1977) based upon a repeating unit cell concept.

Dynamic stiffness and damping properties, which are of great interest in flutter analysis, have been treated in the literature (Material Sciences Corporation 1976, Jones 1975). A review of literature on experimental methods of determining dynamic stiffness and damping properties of a variety of laminate geometries is available (Bert and Clary 1974). Studies of temperature dependence of complex moduli have been reported (Heller, et al. 1975). Viscoelastic considerations are also available (Halpin and Pagano 1968). Refer to Additional Literature, Dynamic Behavior for more references. Moisture and temperature are modeled as alterations to complex moduli (Material Sciences Corporation 1976, Augl and Berger 1977). Jones reviews many vibrational analysis papers.

Moisture and temperature excitations have generally been shown (Whitney 1976, Browning, et al. 1977, Mazzio and Mehan 1977) to be capable of degrading laminate stiffness. The effect is most pronounced in laminates which heavily load the matrix. The change in stiffness anticipated in typical aerospace structure due only to in-service moisture does not appear to be highly problematic in itself. However, hygrothermal excitations combined with flaws, design and manufacturing variables, and in-service loads may have deleterious effects on stiffness.

Supersonic airspeed structural heating and turbine temperatures have led to great desire for matrix materials having less temperature sensitivity. Such matrix materials must be inexpensive to process, and will hopefully be improvements on the present 250°F or 350°F curing temperatures.

Much work concerning improved temperature properties centers around blade and vane development, and therefore sources of such work are much the same as those mentioned in "Impact Strength" concerning FOD research on engines.

Stiffness of Gr fibers in a thermoplastic matrix has been reported (Maximovich 1977). Stiffness of whisker composites has been reported (Lavengood 1972). Hybrid Gl/Gr fiber ratio effects on stiffness have been reported (Kalnin 1972).

Again, as with many other properties, much of the experimental and to some extent analytical stiffness literature is plagued by characteristics which limit its utility to designers. Problems include the tendency towards funding previously funded popular research topics, and deficient, non-standardized reporting procedures. Many designers comment that presentation of plotted results is often lacking. In contrast to such problems, strong analytical capabilities of predicting stiffness have been developed and compared to experimental results (Jones 1975). Long-term stiffness properties with service excitations, including fatigue loads, flaws, and design and manufacturing variables have not been clearly identified in the literature. Much aerospace research is presently pursuing such information.

### Tensile Strength

Tensile strength has received by far the most attention of all parameters in the literature. Many excitations, flaws, and design and manufacturing variables have been addressed. The fact that these experiments are relatively easy to run, and the wide availability of tensile test machines may have much to do with the proliferation of tensile strength literature. Tensile strength is of obvious interest to designers, however, the combinations of excitations, flaws, and design and manufacturing variables treated in the literature often appear to diverge from representative structural situations.

Collections of tensile strength data and/or stress-strain curves are available in several sources (Advanced Composites Design Guide, Third Edition 1973, Jones 1975, Sendeckyj, et al. 1975).

Tensile test coupons are predominantly small thin flat rectangular specimens gripped at each end and pulled axially. A few exceptions are found in the literature, notably a split-D apparatus (Knight 1977) and a pulley and cable apparatus reportedly capable of applying pure moments (Hill and Anderson 1972) both of which are reported to be useful for measuring tensile strength. Some early literature discusses tensile test methodology (Hancock and Swanson 1971) and several papers treat off-axis coupon testing and end constraint concerns (Richards, et al. 1969, Pipes and Cole 1973, Rizzo 1969, Pagano and Halpin 1968, Phoenix 1972). Recent discussion at a composites conference (Bergamo Center 1978) indicated that some variations in test results may be caused by the type of tensile test machine being used. It was mentioned that the phenomenon may be caused by differences in machine energy feeding rates to the specimen. Some literature treats strain rate effects. B/A1 and B/E elevated temperature and high strain rate effects are reported (Meyn 1974), as well as strain rate effects in other materials (Chiao and Moore 1971, Mullin, et al. 1968).

It is apparent from the literature that G1/E tensile strength under a static load is a function of time. This phenomenon is reported as both "stress rupture" and "static fatigue," which is confusing to the uninitiated. Tensile strength vs. log time is shown (Wu and Ruhmann 1975) to be nearly linear except initially. The same authors review various failure criteria. Other research confirms the time-dependent tensile strength of G1/E (Hahn and Kim 1975, Chiao and Moore 1972, Chiao, et al. 1972, Lifshitz and Rotem 1970, Chiao and Moore 1971). A reaction-rate theory has been proposed to explain time dependent behavior (Akay and Saibel 1978). Some early treatment of time dependency effects uses Weibull functions (Halpin, et al. 1970).

G1/E time-dependent behavior has been compared with the same phenomenon in K/E (Chiao, et al. 1974). More recently, an accelerated test technique for predicting K/E lifetime under a given tensile load has been confirmed experimentally (Chiao, et al. 1977). The same authors (Chiao and Moore 1972\*) reported earlier studies of K/E tensile properties, and have compared G1 and Gr fiber strength to Gr and G1 bundle strength (Chiao and Moore 1970).

Some creep analysis of laminates is also found in the literature (Foye 1975, Lou and Schapery 1971).

Many treatments of tensile strength in the presence of holes, notches and slits are available. This body of literature is reviewed in the "Fracture Strength" and "Notch Sensitivity" sections of this report. Gr/E tensile strength influenced by bolt or pin filled holes, with various fastener torques and hygrothermal excitations have been reported (Wilkins 1977).

Also, treatment of the effects of cyclic loading on (residual) tensile strength are found in the literature, and the reader is referred to the "Fatigue Strength" section of this report. Conclusions are difficult to draw, as is mentioned by Lauritis on p. 268 of STP 636.

Reviews of analysis of lamina and laminate tensile strength are readily available in standard textbooks (Jones 1975, Ashton, et al. 1969). Some analyses of transverse tensile loading which may not be found in these textbooks

attempt stress-strain curve predictions using various assumptions (Huang 1973, Adams 1970, Adams 1974). The effects of voids and unbonded fibers are included in an analysis using Mises-Hencky strength criteria (Greszczuk 1975\*). Other efforts include inelastic (Adams 1970) and photo viscoelastic and computer stress analysis (Hackett and Krokosky 1968, Igbal and Krokosky 1970). Unidirectional B/Al and B/E have been treated using elastic-plastic analysis (Lin, et al. 1972). Some efforts towards predicting tensile stress-strain curves analytically are found in the literature (Chou, et al. 1976, Hahn and Tsai 1974\*, Jones and Nelson 1975). Micromechanical analysis of a bead-filled composite which accounts for matrix temperature and strain rate effects is available (Nicolais, et al. 1972).

A statistical method of describing boron fiber tensile strength is available with attention given to long fiber length effects (Phoenix 1975). A review paper of probabilistic models of fiber, fiber bundle, and laminate tensile strength is available (Phoenix 1974). Strength distributions of single filaments have been reported (Larder and Beadle 1975). A "Chain of Bundles" probability model of strength has recently appeared in the literature (Harlow and Phoenix 1978a and 1978b). A host of design and manufacturing variables have been treated, and analysis is available which treats many of them (Adams 1974). Tensile strength as influenced by delaminations, edges, and stacking sequence is treated in the literature (Reifsnider, et al. 1977\*). Stacking sequence (Pagano and Pipes 1971) and B/Al processing (Joseph, et al. 1968) are also discussed. Transverse tensile strength of C/C as influenced by voids is found in the literature (Brassell, et al. 1975).

Improvements of up to 10% in average strength and 50% reduction in standard deviation have been demonstrated by prestressing prepregs in tension or bending (Mills, et al. 1975).

The effects of B/E reinforcements on Al and Ti metals tensile strength have been reported (Herakovich, et al. 1974).

Many papers discuss NDE during tensile loading, and some specifically focus upon NDE. Acoustic emission is treated in several such papers (Hennecke and Herring 1975, Hamstad and Chiao 1975, Pipes, et al. 1977).

Moisture and temperature excitations have been treated in the literature, both independently and combined. In general, the literature indicates that these effects are most pronounced on laminates which derive much of their strength from the matrix (Springer 1976, Browning and Hartness 1974, Shen and Springer 1977\*, Whitney 1976, Hofer, et al. 1977, Wilkins 1977, Sun and Roderick 1977, McHenry and Tressler 1975, Ishai and Mazor 1975, Unknown NBS 1978, Berman 1975, Mazzio and Mehan 1977, Pagano and Hahn 1977). First ply failure in Gr/E is treated as a function of moisture content (Pagano 1976). Others argue that moisture absorption may cause swelling which counteracts residual tensile stresses in the matrix (Pagano and Hahn 1977). Residual stresses are treated elsewhere in the literature (Sun and Roderick 1977), and are considered important in reducing allowable strain to first ply failure (Chamis Lecture at GWU shortcourse #397, 1977). Qualitative discussion of matrix and sometimes interface effects is often encountered, including lowering of  $T_g$ , glass transition temperature. However, very little quantitative treatment of resin mechanical properties is found. One might, for example, study resin toughness or shear strength as a function of moisture, temperature, and other parameters if indeed the resin appears to be the weak link. Moisture cycling excitation effects on Gr/E tensile strength and fatigue strength have been reported (Lunderno and Thor).

Thermal cycling effects on Gr/E tensile strength have been reported (Mazzio and Mehan 1977), and thermal cycling in hard vacuum data is also available (Berman 1975). Cryogenic (Kasen 1975, Unknown NBS 1978) and liquid nitrogen (Chiao, et al. 1975) temperatures have shown mild effects on Gr/E, and it would appear that elevated temperatures tend to degrade epoxy matrix and hence laminate properties, whereas dropped temperatures generally do not cause degradation. Such behavior promises to be an advantage of composites in machinery exposed to low temperatures, wherein many metals suffer significant loss in fracture toughness.

Laser impingement upon G/E has been shown to have an effect upon tensile strength akin to drilled holes (Kibler 1976, Kibler, et al. 1975). The effects of impacts of various types, sizes, and velocities upon tensile strength have been studied (Walter, et al. 1977, Reed and Schuster 1970, Carlisle, et al. 1975). Buckling has been shown to degrade tensile strength of Gr/E (Bhatia 1976). Benzene immersion has been shown to cause a reduced failure envelope for Gl/E (Wu and Ruhman 1975).

Designers seem to be hungriest for research on properties other than tensile strength, since service experience has generally shown other material properties to degrade more easily or at quicker rates. In light of this, future research emphasizing other material properties is more appropriate. Continuing research in tensile strength should address those combinations of flaws, excitations, and design and manufacturing variables which mimic most service.

### Thermal Conductivity

A rather small amount of published literature addresses thermal conductivity of composites. Thermal conductivity of popular Gr fibers has been shown (Kalnin 1975) to correlate with fiber microstructure and hence with other fiber properties such as modulus and electrical conductivity. In unidirectional composites, the ratios of longitudinal to transverse conductivity are typically (Noor 1977):

	$K_{11}/K_{22}$
B/A1	1
B/E	3
HT-S Gr/E	16
GY-70 Gr/E	16

Rule of mixtures expressions are available for thermal conductivity in the longitudinal (Chamis 1977, Kalnin 1975) and transverse (Chamis 1977) directions of unidirectional composites:

$$K_{l11} = V_f K_{f11} + V_m K_m \quad (1)$$

$$K_{l22} = (1 - \sqrt{V_f}) K_m + \frac{K_m}{1 - \sqrt{V_f} (1 - K_m/K_{f22})} \quad (2)$$

where:

- $K_{l11}$  = laminate thermal conductivity in (longitudinal) fiber direction
- $K_{l22}$  = laminate thermal conductivity intratransverse direction
- $K_{f11}$  = fiber conductivity in longitudinal direction
- $K_{f22}$  = fiber conductivity in transverse direction
- $K_m$  = matrix thermal conductivity
- $V_m$  = volume fraction of matrix
- $V_f$  = volume fraction of fiber

Void content may also be taken into account (Chamis 1977), by modifying the expression for  $K_m$ . Since  $K_{f11}$  of Gr is usually at least 20 times  $K_m$  of epoxy, the second term of (eq. 1) may often be neglected (Kalnin 1975) for Gr/E. An expres-



sion for laminate heat capacity is also available (Chamis 1977) based upon rule of mixtures.

Analysis providing bounds on transverse thermal conductivity is available (Elsayed and McCoy 1973, Sanford and Silnutzer 1971). Analysis and experimental data for transverse and normal conductivity in unidirectional composites has been reported (Springer and Tsai 1967), and has received some commentary in the literature (Zinsmeister 1970). Analysis and data for GI/Polystyrene is available (Sundstrom and Chen 1970). Analysis based upon variational principles has been reported (Donea 1972). The method of long waves has been used to obtain thermal conductivity in three orthotropic directions (Behrens 1968). A conductivity problem having a round heat source on one face of a laminated plate has been treated (Yovanovich 1970).

Some data showing dependence of thermal conductivity upon cryogenic temperatures is available (Kasen 1975) for Gr/E and B/E over a temperature range from 77° to 295°. This interest stems from the fact that many composites have much lower thermal conductivities than do metals.

Thermal conductivity of Graphite composites for purposes of sliding bearing applications have been reported (Brown and Blackstone 1974).

More recently the effects of cracks on thermal conductivity have been treated (Hasselman 1978).

The ranking of ablative performance of several C/C composites is correctly predicted (Guess and Bulter 1975) by a thermal shock factor model which takes into account thermal conductivity.

## A NOTE FROM THE SHOP

"The state-of-the-art seems to be that the fibers have become so strong and stiff that the resins can't hang on to them anymore, and that the resins have now become the weak link."

(personal communication with J. M. Brown)

## DISCUSSION OF RESULTS

It is recognized that recommended priorities and supporting arguments for composite research are specific to each user industry. We have chosen to address the aerospace industry in particular. The aerospace community needs efficient methods for designing, manufacturing, and assuring integrity of advanced reinforced fibrous composite structures, particularly primary structure. We believe that structural safety, that is, failure prevention, should be assured to an extent at least comparable to, if not better than, metals if composites are to become generally applicable. Furthermore, composites must remain cost-competitive. Specific recommendations for future research in advanced fibrous composite materials which will best serve the aerospace community derive from our state-of-the-art review of aerospace service experience and literature.

We anticipate that the means for (1) designing and (2) assuring structural integrity cannot be wholly delineated until a broader experience base is available, as is the case for metals. Moreover, it is anticipated that these two tasks, in the near future, will have to be developed and treated more separately and distinctly than they are in metals. This follows from state-of-the-art composite design processes being less capable of accurately predicting material behavior and anticipating damage morphology than are current metal design processes. As the capabilities of predicting the effects of flaws, excitations, and material design and manufacturing parameters on material properties and damage become more accurate, then the design process can go further toward accomplishing reliability assurance. The two tasks may then become more integrated, as they are now for metals.

A state-of-the-art review of composites shows that there already are a significant number of successful applications of composites to both primary and secondary structures, now in service. Some design and reliability assurance philosophies and techniques have evolved. Despite arguments that success has often been achieved by choosing situations where composites are relatively easy to use in a direct manner or where the risk of failure is not punitive, more challenging choices for use of composites have emerged as well and, cautiously, the aerospace industry is moving to the point where increased amounts of primary structure may be made of composites. It is logical to proceed by using state-of-the-art applications and experiences as milestones and pointers towards developing effective methods for designing and assuring structural reliability. Still, a great deal of research is needed, and specific items are described in the following sections.

## CONCLUSIONS (RECOMMENDATIONS)

### Design Methodologies

Design methodologies must be developed and extended to account for those effects which may cause damage\* to actual structures. Such effects include material design and manufacturing variables\*, flaws\*, and excitations\*. The design methodologies must explicitly recognize that material design, structural design, and manufacturing processes are intimately related. Development should proceed by recognizing that design and reliability assurance may be most effectively treated as two somewhat separate tasks.

It is apparent that these methodologies must rely upon a set of predictive models, some of which are embryonic and others have yet to be developed, fully. Moreover, the design methodologies must draw from a common material data base, now only in its infancy. This approach will replace more ad hoc methods, e.g., full scale testing, and is obviously to become highly cost effective.

The type of procedures which will best serve the aerospace community for design purposes are a mixture of analytical, semiempirical, and empirical. The design tools or models which are most useful are both micromechanical and macromechanical. However, micromechanical design models are viewed as being accurate in only the simplest of conditions, and they are useful for providing the designer with a material design rationale. That is, micromechanical analytical models treat simple configurations, limited design and manufacturing variables and, typically, few combinations of excitation. When the complexities of 3-D structure having inherent design and manufacturing variables under multiple excitations are incorporated, it has been found that micromechanical analytical models are deficient in predicting damage.

\*See Appendix B

The inaccuracies of micromechanical models seem to arise from the following sources:

1. Simplified material representation (linearity, for example).
2. Few rather than many failure modes considered simultaneously.

They may offer insight into local modes of damage accumulation or growth. Therefore, it should be recognized that the utility of micromechanical modeling is best limited to material design rationale but, for the purpose of establishing more accurate and effective structural design methodology, macromechanical modeling is the approach more likely to pay off.

*It is therefore recommended that development of semiempirical or empirical macromechanical models proceed with high priority.* Analytical models are now reasonably mature and, at best, developed further to offer only incremental insight. These models should be useful tools for relating actual material design and manufacturing variables, excitations, and damage data, i.e., to create empirical and semi-empirical models. They are also expected to demand a more complete base of experimental data. *These models are needed to develop acceptability levels for quality assurance during manufacture and reliability assurance in service.*

The service experience collectively demonstrates that certain items are of immediate importance. Development of predictive models should therefore reflect the following priorities:

- reduced need for wizards, composite engineers with extraordinary expertise
- effects of delaminations and other flaws on compressive fatigue strength, with hygrothermal excitation
- joint strength, both bonded and fastened

- post buckling fatigue and other behavior pertinent to fuselage design
- crashworthiness
- repairability
- real spectrum loading *vs* accelerated test techniques
- 3-D effects
- inexpensive and rapid methods of testing

Moreover, these models either are now or will provide the basis for failure analysis, in the event of damage or structure failure. The problem is compounded by the likelihood of multiple mechanisms of damage operating simultaneously at different sites, and of multiple failure origins. *Nonetheless, the capability for failure analysis is required these methods should accommodate a wide range of damage mechanisms so that the most probable source of failure is clearly identified.*

### Material Characterization

*It is of utmost importance to develop a broad, clearly reported, repeatable data base in support of design models and reliability assurance models. The data base should provide the following information:*

- *Laminate Properties*, as influenced by excitations, flaws, and material design and manufacturing variables. The combinations of flaws, excitations, and material design and manufacturing parameters which are currently of highest research priority are:
  - (a) toughness against delamination growth;
  - (b) compressive fatigue strength, particularly after FOD (impact) excitation; and
  - (c) joint strength.
- *Lamina Properties*, as influenced by excitations, flaws, and material design and manufacturing variables.
- *Correlation of constituent with lamina and/or laminate properties* is needed to quantify the effects, if any, that changes in constituent (matrix, fiber) properties have upon the final laminate. This will indicate laminate properties which may be effectively improved by changes in constituent properties. It will also serve to flag constituent properties which may be worthy of inclusion in semi-empirical models. For example, some correlation between resin toughness and laminate fracture strength against delamination growth may exist.
- *Constituent material properties*, that is, changes in properties of matrix materials and fibers at the constituent level which may be used to improve laminate properties. Whenever useful correlation is shown to exist, data is needed indicating how to change constituent



properties. For example, experience indicates that resin toughness might correlate to some extent with crosslink density. Unfortunately, in-situ rather than neat\*, measurements may be required. This research may well proceed more effectively with increased interaction of chemists and mechanists.

### Improved Material Properties

Improvements in various material properties are desired. The specific priority is dependent upon the individual application; we list here those of greatest concern.

- Improved toughness against delamination growth
- Lessened hygrothermal sensitivity
- Higher stiffness without strength degradation
- Higher strength
- Improvements associated with electrical conductivity\*\* are:

1. Improved lightning protection, where applicable.

(This research may interact highly with manufacturing techniques as a solution, but the problem arises as an inherent material property.

2. Improved system integration properties where antennae, grounding, shielding, and other electrical characteristics have been problematic.

\*Neat is a term referring to properties of a constituent in bulk rather than in the final product form.

\*\*Lessened hazard due to the conductivity and mobility of fibers after a fire or during manufacture is now under research aimed at assessing the potential hazard and developing ways to reduce the hazard. Research funding on this problem looks quite adequate.

## Testing and Reporting Procedures

*In order for experimental data to be wholly useable and repeatable, comprehensive testing and reporting procedures must be developed which provide information useful throughout the composites community.*

**Standard Test Methods:** Much confusion could be avoided by developing standardized test methods. Development of the test methods must allow for the potential differences in failure modes typically forced in distinct applications and must account for edge, size, and other effects where necessary. For example, standard compression test methods must allow for both variations and combinations of edge support conditions so that each designer has data useful and applicable to his application. The implication is that research may well indicate the need for a family of standard compression tests. Much the same may be said concerning fatigue and shear testing. Some testing may be done on the lamina level if design models are able to show correlation to more complex situations.

**Reporting Procedures:** In order for reports to be useful throughout the composites community, it is necessary to have complete reporting of all those parameters which might have affected the data. It is expected that this will in general result in a rather tedious and somewhat lengthy testing report compared to metals, inasmuch as composite material properties and damage are sensitive to more variables. In particular, the following types of information should be considered for inclusion in reporting procedures:

1. Material design and manufacturing variables, flaws, and measurement methods (as defined in the Vocabulary - specimen and fabrication description is part of this).
2. Excitation and its measurement methods.

3. Damage modes and failures, reporting measurements clearly, including photographs as appropriate. Note that damage as we have defined it does not necessarily constitute failure. A perusal of the literature will show that words like damage, failure, flaw, environmental effects, are used in many different ways.
4. Unreduced data in some instances, such as raw printout or plots of structural response.

(We note that ASTM is already actively pursuing standardization of composite testing and reporting procedures. Activities currently appear to be centered within Committees D-30 on High Modulus Fibers and Their Composites, E-9.03 on Fatigue of Composites, and E-9.07 on Environmental Effects on Fatigue. The aerospace community is well represented and active in these activities. During 1978, ASTM efforts included surveys concerning industrial methods of testing composite properties, including impact, compression, fracture, fatigue, and shear testing. Also, efforts were underway to define terms and develop nomenclature concerning composites. It appears that standards, for the most part, are currently unavailable for composite testing and reporting, and are in their infancy. Fiber content may be determined using ASTM D3171, Test for Fiber Content of Reinforced Resin Composites, and void content may be determined using ASTM D2734 Test for Void Content of Reinforced Plastics (Kalnin 1975). ASTM activities also include conferences which have in the past generated Special Technical Publications (STP 452, 460, 497, 521, 524, 546, 568, 569, 580, 593, 602, 617, and 636) pertinent to composites. Development of testing and reporting procedures may therefore evolve most expediently within the continuing context of ASTM; *the point here is that these efforts must continue as a matter of high priority and that substantiating research should be supported where pertinent.*)

## Manufacturing Methods

Manufacturing methods for composites are multistep processes which may involve different raw materials; various heat, pressure, and cure cycles; various molding and heating techniques; and equipment. Unlike metals, the end result including the material properties of the fabricated structure is highly dependent upon processing and even upon prepreg properties.

*Design methodology must therefore account for the manufacturing and processing techniques.* A multitude of tough problems have been encountered in the accumulated composite manufacturing experience including:

1. Thermal expansion mismatch with dies. The problem is that large Gr/E parts of near-zero thermal expansion may be crushed when metal molds cool down.
2. Void content, resin/fiber ratio, and deficiencies are tough to control.
3. Joining methods are not a clearly developed technology, and bonded joints have given much trouble.

Some of these problems persist in many shops, while others have solved the issues after considerable expense. Technology transfer could conceivably expedite development of more effective manufacturing methods; it must be recognized, however, that much of this kind of information is regarded as proprietary.

In addition, a problem concerning manufacturing quality assurance and FAA regulation has developed. *Interest should be in regulating the integrity of the structure which results from manufacture.* However, instead of directly dealing with material properties of the finally fabricated structure, the regulations have attempted to indirectly regulate the result by placing constraints upon manufacturing processes. Much difficulty has already occurred in this respect. This entanglement of regulations with manufacture and design is seemingly

unnecessary and can limit the designer's freedom of choice to an extent where less efficient and more costly structure results. It may also force rejection and scrapping of non-standard parts which do not meet regulations but whose integrity could otherwise be assured. *Therefore, research in manufacturing should address development of cheaper automated processes which produce acceptable materials.*

### Development of Quality Assurance

As an integral part of design and methodologies, material and structural properties must be quality assured, and experience indicates that Q/A will be required at several stages of manufacture. However, the primary focus of quality assurance development should be upon the end result of manufacture.

- *Evaluation of prepreg properties upon receipt from suppliers and immediately before manufacturing use is necessary, based upon past prepreg inconsistency and storage aging problems. Experience shows that it is insufficient to expect repeatability of laminate properties by describing the same old process and by saying that "Brand X-1234 prepreg was used." The current proprietary nature of the prepreg business combined with the inability to Q/A prepregs upon receipt results in use of prepregs with inconsistent properties. This impedes the art. Prepreg Q/A should help alleviate this problem.*
- *Structural Evaluation. Methodologies must be developed to assess the quality and consistency of the manufactured structure. It may be most efficient and economical if the methodologies are applicable at critical stages of the processing so that problems are traceable, but it is desirable that the evaluation method*

should allow manufacturing processes to be internally, not externally, regulated with the end result being important.

### Reliability Assurance and NDE Development

Reliability assurance has proceeded most successfully by:

- identifying the types, amounts, and combinations of damage which are critical to the structure, and
- always keeping damage accumulation less than the identified critical amount during flight, via periodic NDE.

The first task has been and will most likely continue to be unique to each structure, since loads and geometries will vary, sometimes considerably. Thus each potential application of composites must be evaluated in terms of its special blend of damage accumulation modes. No simple, thorough, standardized list is likely to be identified in the near future. As the experience base grows - in parallel to the present situation for metals - standard design configurations, design methods, and methods of assuring integrity may develop as potential failure modes become more easily anticipated.

*Therefore, development of periodic in-service inspection, via NDE, proof test, or any other reasonable means, is essential so that performance and integrity which are to some extent forecast at the point of design may be confirmed on a continuing basis. Unlike inspection of metallic structures, however, the modes of flaws to which inspection must be sensitive are far more extensive; thus NDE itself becomes more complex.*

*Repair methods must be developed which enable damaged composites to remain cost-competitive with comparable metal structure. Both large and small repair areas must be addressed in anticipation of NDE showing service deterioration of various forms and sizes. For smaller sizes, it is desirable that repairs be made in the field rather than returning the part to the factory. Repairs must*

restore integrity to the part. The development of repair methodologies is strongly associated with development and choice of joining methods in that patches must be joined to parent structure.

### Development of NDE

*In order to effectively support design and reliability assurance methodology, reliable, thorough, and cost effective NDE must be developed. Such NDE must detect and quantify those flaws and design and manufacturing variables which are identified as critical to the lifetime of the part. Some NDE may therefore be a supporting technique for Q/A.*

Service evaluation NDE methodologies are needed to detect and quantify the damage or lack of damage which a part has accumulated during service before failure occurs. If damage is measured directly, i.e., using a method such as periodic proof tests, then the need for NDE during service may be precluded. However, experience shows that the potential of multiple failure modes causes problems with such a direct excitation damage measurement methodology. Furthermore, it appears that it would be structurally inefficient to design around this problem.

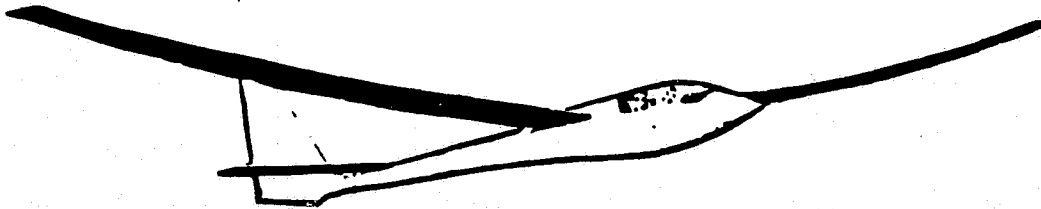
It then seems appropriate to primarily measure or infer damage indirectly. By knowing the status and extent of pertinent excitations and design and manufacturing variables and using developed failure prediction methods, the desirable reliability could then be ascertained with confidence.

Already, service experience has indicated a need for NDE of FOD-caused excitations (cracks, delaminations, fractured fibers, debonds, etc.). *The NDE detectability of all identifiable possible causes of damage must be assessed, with the goal of indicating those causes which are not efficiently detected using current methods.*

APPENDIX A  
AEROSPACE APPLICATIONS

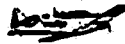
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ASW-17 Sailplane Fuselage Tail Cone, Gr/E, a production option for higher stiffness, 3-4% lower empty weight, reinforced fuselage; allowing installation of a drouge chute, increased flutter speed (Hodges 1977).



(photo adapted from Hodges 1977)

AT-6 Outer Wing Panels, Wright-Patterson AFB, Gl/PE, heavier than comparable aluminum wing panels, but ultimate load was increased, 13% increase in strength/weight was shown by test, problems with weathering and temperature properties of PE resin remained (Hackman 1973).



(photo adapted from Hackman 1973)

A1E Vertical Stabilizer, Grumman Aircraft Corporation, reported as first production application of Gl/E in military aircraft, required for electronic purposes (Hackman 1973)

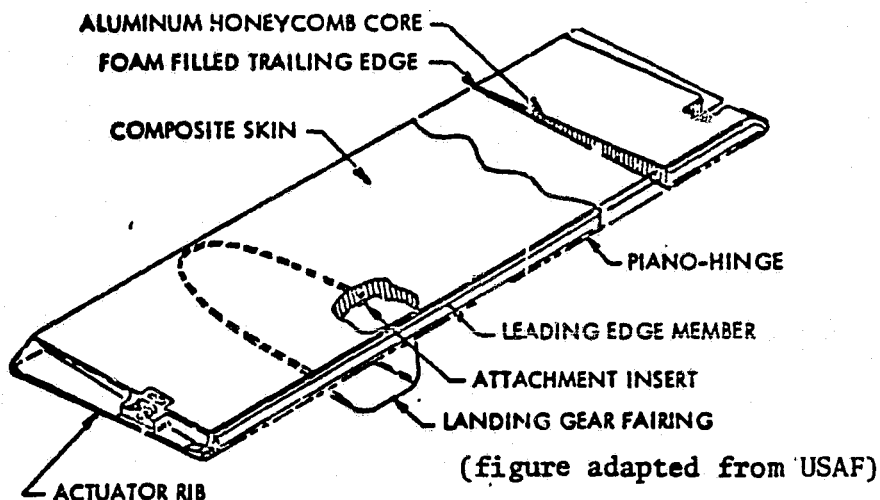


(photo adapted from Hackman 1973)

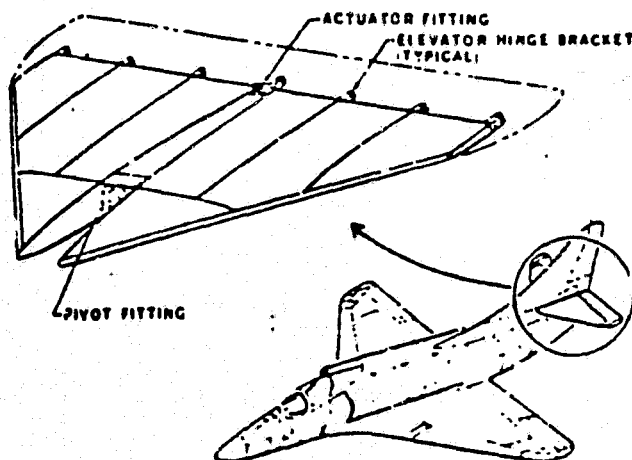


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A-4 (Skyhawk) Wing Landing Flap, McDonnell Douglas/U.S. Navy, two composite versions have been constructed, B and Gr, B version has full depth honeycomb core with metal ribs and fittings and separate upper and lower skins, weighs 27% less than Al part and has 70% fewer parts, Gr version has honeycomb core with Gr rib and continuous skin wrapped around trailing edge, weighs 39% less than Al part and contains 89% fewer parts (Salkind 1976, Advanced Composites Design Guide, Third Edition 1973 Vol. V, Section 5.1.2.29).



A-4 (Skyhawk) Horizontal Stabilizer, McDonnell Douglas/U.S. Navy, Gr/E skins adhesively bonded and mechanically fastened to Gr/E and honeycomb shear web substructure, 28% lighter than metal (Salkind 1976, Advanced Composites Design Guide, Third Edition 1973 Vol. V, Section 5.1.2.3.3).



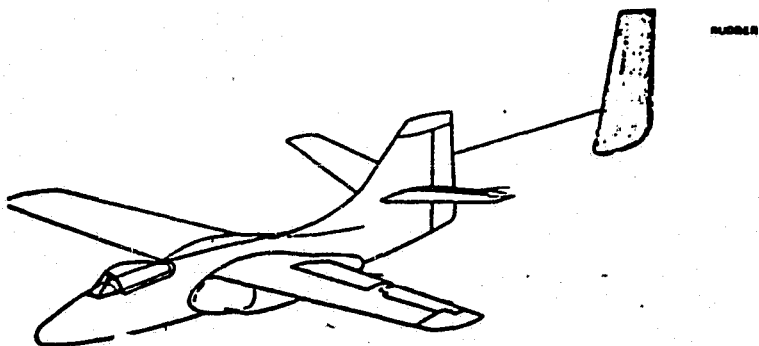
(figure adapted from USAF)

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A-7 (Corsair) Outer Wing Skin, Navy craft with hinged wings for storage on carrier operation, Gr/E on 8 in-service craft, regularly inspected (personal communication with Frank Fecek).

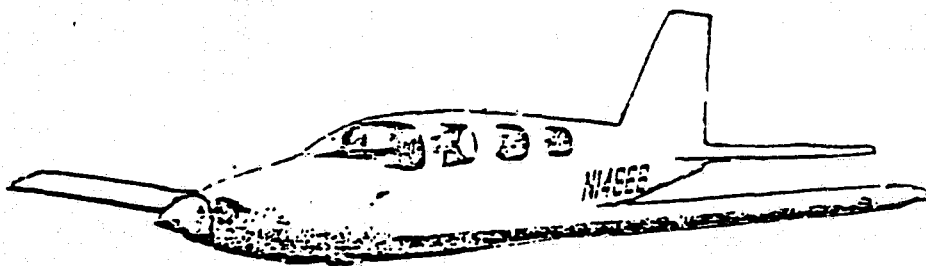
A-7 (Corsair) Speed Brake, Gr/E (Advanced Composites Design Guide, Third Edition 1973 Vol. V, Section 5.1.2.4.4).

A-9A Rudder, USAF, Northrop Corporation, chosen to demonstrate low cost manufacturability, was reportedly flight tested (Advanced Composites Design Guide, Third Edition 1973, Vol. V, Section 5.1.2.1).



(figure adapted from USAF)

Bellanca Skyrocket, Bellanca Aircraft Engineering, Inc., G1/E skins on Al honeycomb core in some parts (Bellanca 1977).



(figure adapted from Bellanca 1977)

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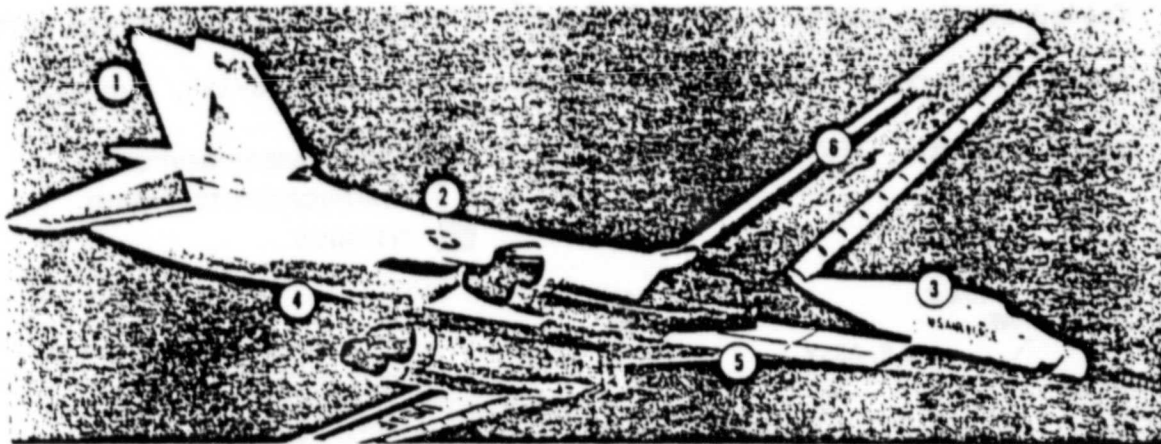
BQM-34E Drone Wings, Navy, Gr/E skins on Al honeycomb core with titanium root carry through, four wings installed and monitored periodically, foreign object damage has been successfully repaired (personal communication with Dr. M. S. Rosenfeld).

BT-15 (Vultee) Aft Fuselage, USAF Wright Air Development Center, GI/PE skins on balsa wood core, reportedly fabricated 1943 (Hackman 1973).



(figure adapted from Hackman 1973)

B1 Avionics Access Door, secondary, USAF (Rockwell International Brochure NA-78-195, Altman and Olsen 1977).



1. VERTICAL BOX



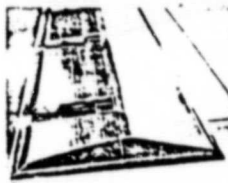
2. DORSAL LONGERON



3. AVIONICS ACCESS DOOR



4. AFT ELECTRONICS BAY DOOR



5. WEAPONS BAY DOOR



6. SPOILER

(figures adapted from Rockwell International 1978)

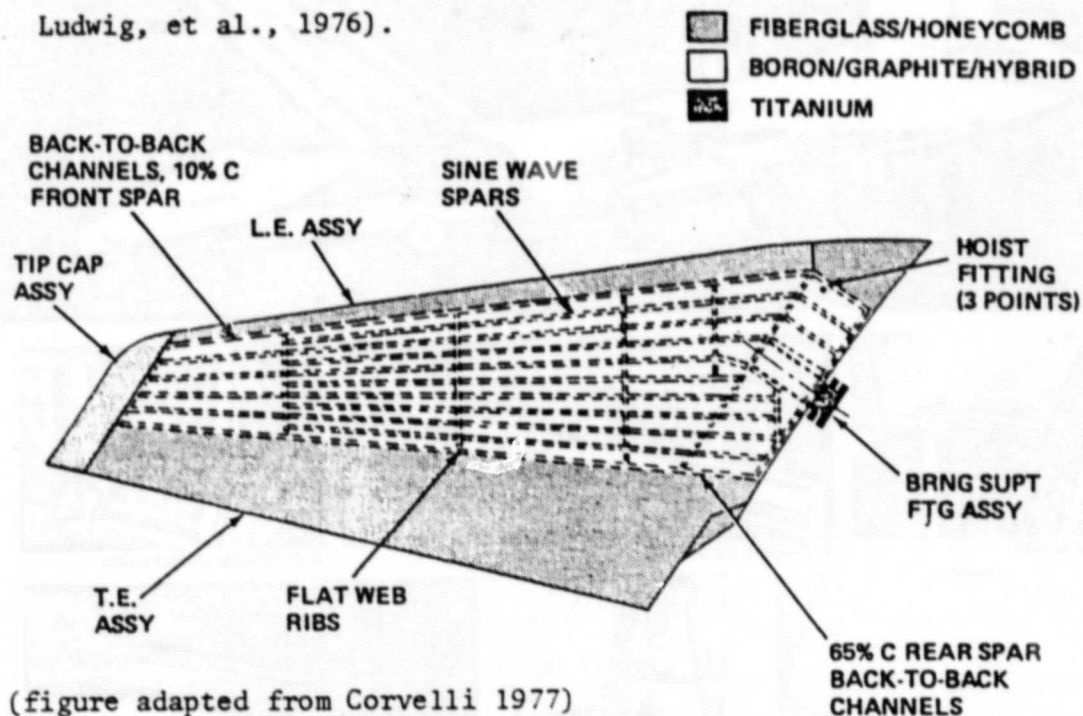
B1 Flap, secondary, USAF (Rockwell International Brochure NA-78-195, Altman and Olsen 1977).

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B1 Horizontal Stabilizer, primary, USAF, Grumman, 1974, each stabilizer weighs 1294 kg (2846 lb) and has .6x9 m (2x30 ft) span with 5m (17 ft) chord at root, materials used:

Graphite/Epoxy	48.6%
Boron/Epoxy	7.5%
Glass/Epoxy	5.2%
Titanium	13.3%
Aluminum	18.9%
Misc	6.9%
	100.0%

very extensive tests including flaws, static, fatigue and hygrothermal excitations, qualify this part for production, metal equivalent might well have failed similar tests, saves 15% weight, 17% lower cost, 40% fewer parts, 60% fewer fasteners (Rockwell International Brochure NA-78-195, Corvelli 1977, Ludwig, et al., 1976).



B1 Longeron, USAF, B/E bonded to five major metal fuselage longerons saves more than 455 kg (1000 lb), some joints with thermal residual stresses debonded and

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were repaired (Dickerson, et al. 1974, Rockwell International Brochure NA-78-195, Advanced Composites Design Guide, Third Edition 1973 Vol. V, Section 5.1.2.4.2).

B1 Slat, secondary, USAF (Rockwell International Brochure NA-78-195, Altman and Olsen 1977).

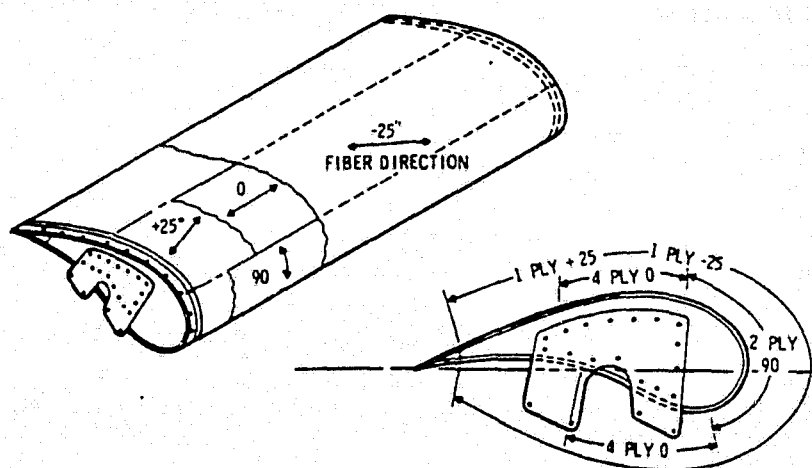
B1 Vertical Stabilizer, primary, USAF, International Rockwell, subsequent and similar development to horizontal stabilizer, however, systems integration problem may have led to a decision not to include it as a production item, much design effort required to design root joint area where transition to metal occurs (Altman and Olsen 1977, Corvelli 1977, Rockwell Brochure NA-78-195).

B1 Weapons Bay Door, secondary, USAF, debond from a manufacturing test showed up in tests (Altman and Olsen 1977, Rockwell International Brochure NA-78-195).

B17 (Black Widow) Radome (Hackman 1973).

B-29 Radome (Hackman 1973).

B-707 Foreflap, secondary, Boeing Commercial Airplane Company, monocoque structure with B/E skins on .6 cm (1/4 inch) depth aluminum honeycomb core, titanium end fittings, 25% weight savings, two have been in service since 1970 and accumulated greater than 22,000 flight hours (Salkind 1976, June and Lager 1973).

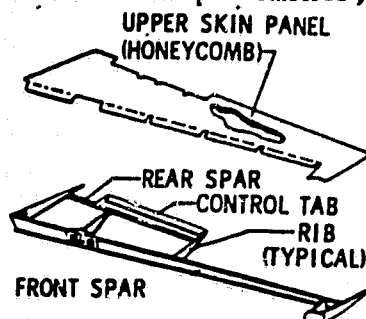
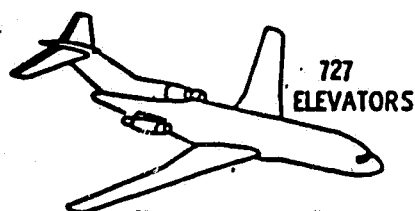


(figure adapted from June and Lager 1973)

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B707 Radome, secondary, Boeing Commercial Airplane Company, in service 1955  
(Thompson 1977).

B-727 Elevator, Boeing Commercial Airplane Company, NASA, developing Gr/E thin  
skins on honeycomb core, subcomponent tests are planned to verify the design,  
design is dictated primarily by stiffness requirements, currently in tooling  
(Vosteen 1978).

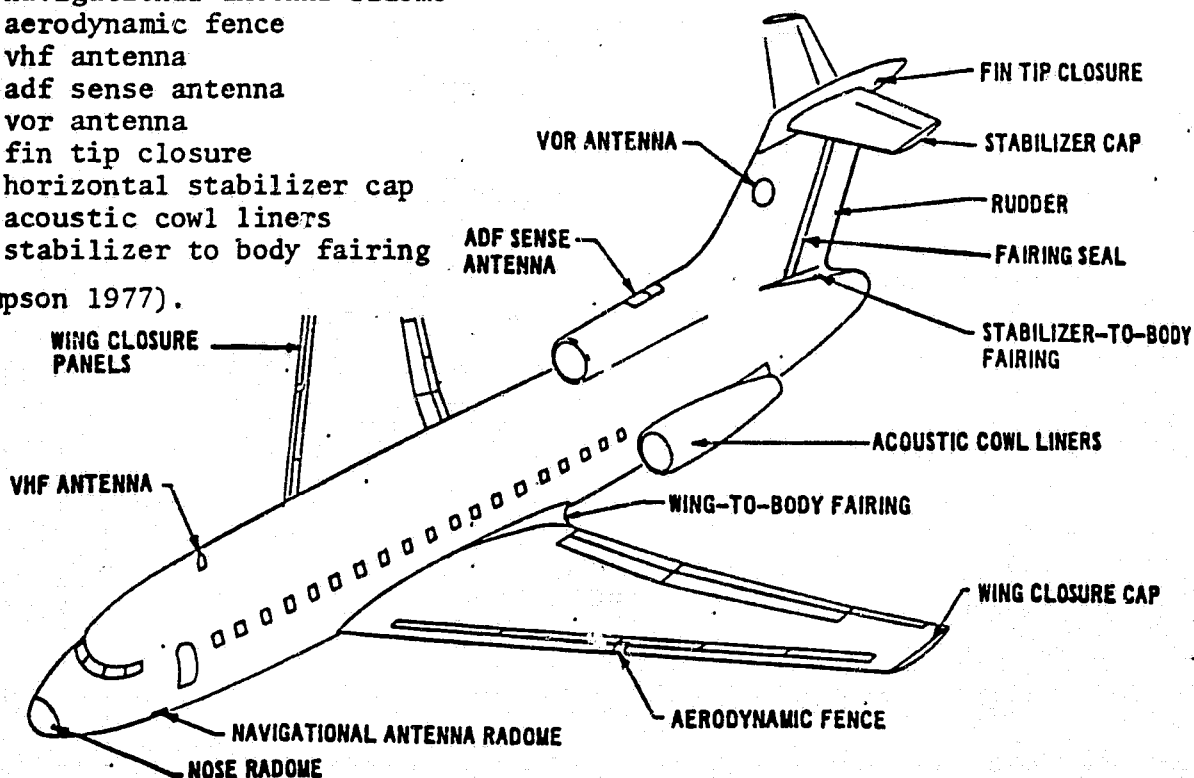


(figures adapted from  
Vosteen 1978)

B-727 Miscellaneous Parts, Boeing Commercial Airplane Company, apparently many  
applications of composites may have existed since 1960 totaling about  $167 \text{ m}^2$   
( $1800 \text{ ft}^2$ ) airplane:

radome (nose)  
wing closure panels  
wing-to-body fairings  
rudder control surfaces  
navigational antenna radome  
aerodynamic fence  
vhf antenna  
adf sense antenna  
vor antenna  
fin tip closure  
horizontal stabilizer cap  
acoustic cowl liners  
stabilizer to body fairing

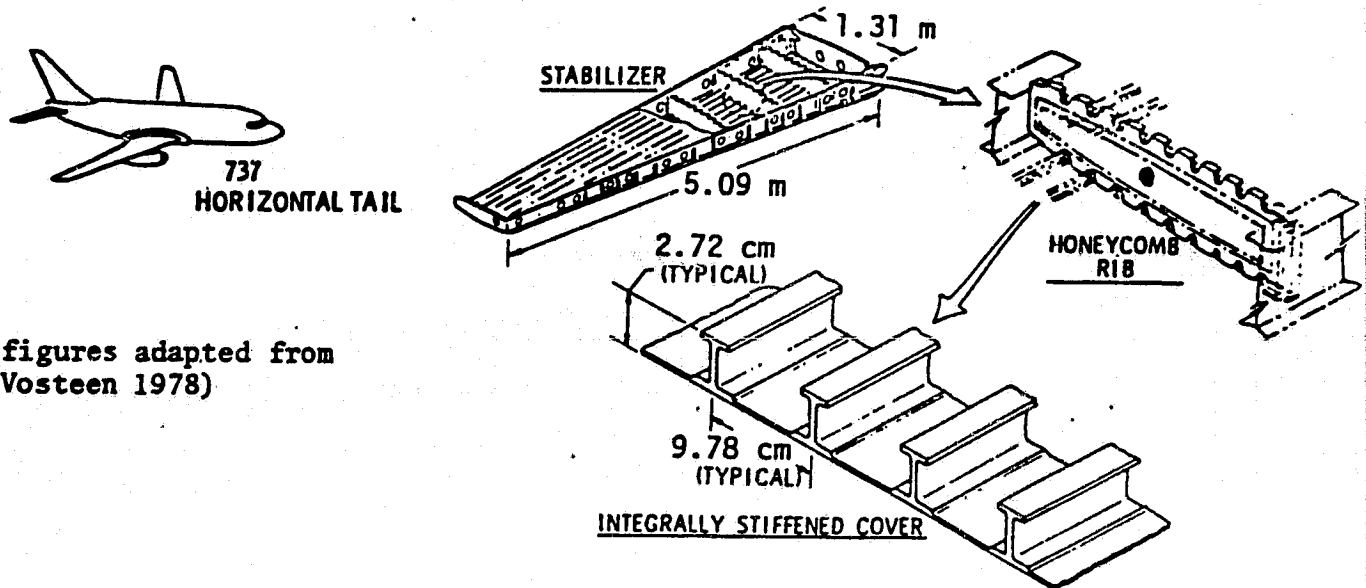
(Thompson 1977).



(figure adapted from Thompson 1977)

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B-737 Horizontal Stabilizer, primary, Boeing Commercial Airplane Company, being developed, ground testing is done, certification flight testing scheduled for March-April 1980, uses an integrally stiffened skin, inspar ribs use honeycomb sandwich webs, titanium reinforcement straps bonded and bolted to built-up spar chords, all load is transferred to lug attaching points to enable these parts to directly replace metal counterparts, 29.7% lighter, total weight LH and RE is 166 kg (365 lb) (Vosteen 1978).



(figures adapted from  
Vosteen 1978)

B-737 Miscellaneous Parts, Boeing Commercial Airplane Company, apparently many applications of composites to parts may have existed since 1965 totaling about 279 m<sup>2</sup> (3000 ft<sup>2</sup>)/airplane; including:

Radome (nose)  
wing-to-body fairing  
flap track and screw fairings  
empennage control surfaces  
wing closure panels  
entry doors  
dorsal fin

vertical stabilizer to body fairing  
tail cone  
wheel well fairings  
aerodynamic fence  
trailing edge panels  
elevator control tab  
horizontal stabilizer cap

(Thompson 1977).

B-737 Spoilers, Boeing Commercial Airplane Company, 139 spoilers have been fabricated, of which 108 have been installed since 1973 in sets of four on 27 aircraft operated by six airlines

21 3048 3/11/78  
 171410 171410

	<u>Aircraft</u>	<u>Spoilers</u>
Aloha	4	16
Lufthansa	6	24
New Zealand National	4	16
Piedmont	8	32
Frontier	2	6
VASP	4	16

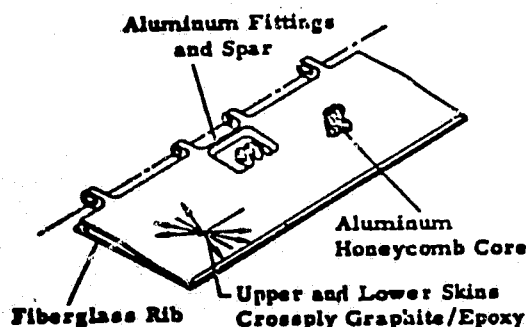
56x130 cm (22x52 inches), 6 kg (13.2 lb) mass, in the first series, 3 types materials of Gr/E skins bonded to full-depth aluminum honeycomb core with metal leading edge spar and hinge fittings, 35% composite gives 17% lighter weight, by May 1978 these parts had accumulated about four years of flight, totaling 921,600 flight hours: program includes annual removal of 6 spoilers, 2 each of the 3 material systems randomly, intensive inspections and destructive testing of residual strength, results show no systematic deleterious trend in strength tests, a few cases of exfoliation corrosion of aluminum parts have been reported and attributed to insufficient edge sealant, one has had an upper skin blister, due to inadvertent installation of larger actuator rod-end, some have been damaged in maintenance hanger area service, successfully repaired at Boeing, and returned to service, no failures have occurred.

Twelve all-composite spoilers were developed and placed in service on six of the previous aircraft which had G1/E honeycomb, molded chopped Gr hinge fittings, close out ribs, and LE spars, and Gr/polysulfone skins, 20% lighter, after 6 months of service the two spoilers on Aloha Airlines developed delaminations on the lower surface skin directly aft of the actuator fitting, within one month a third spoiler was starting to delimitate, it was therefore decided to remove all remaining polysulfone spoilers from service because service excitation of phosphate-ester based hydraulic fluid resulted in skin delamination near the actuator fitting, material testing after boiling specimens in the hydraulic fluid did not identify this potential problem, excellent recordkeeping, follow up,



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and reporting has been done on B737 spoilers (Thompson 1977, Vosteen 1978, Salkind 1976, Hardrath 1977B, Pride 1978, Stoecklin 1977, Advanced Composites Design Guide, Third Edition 1973 Vol. V, Section 5.1.2.2.5).



(figure adapted from USAF)

B-747 Floor Panels, 25 aircraft have varying amounts of British-made panels in the galleys and passenger compartments, they desire to reduce the corrosion problem in and around galleys and laboratories, women's high spiked heels degraded the laminate, the problem was not really solved; fashion changes resulted in less severe loading (personal communication with R. A. Pride).

B-747 Miscellaneous Parts, Boeing Commercial Airplane Company, apparently many applications of composites may have existed since 1967 totaling about  $937 \text{ m}^2$  ( $10,000 \text{ ft}^2$ )/airplane:

- Radome
- wing-to-body fairings
- wing panels
- wing trailing edge panels
- empennage control surfaces
- flap track fairings
- wing closure panel
- apu inlet duct

(Thompson 1977).

B-747 Slats (leading edge), G1/E of variable camber, no chronic problems have been reported (personal communication with Boeing).

B-757 Miscellaneous Parts (personal communication with Boeing).

B-767 Miscellaneous Parts, a long firm list of items to be made from composites, and an additional list of possibilities, all control surfaces except for flaps are candidates (personal communication with Boeing).

#### Carbon (Graphite) Fiber Incidents

The occurrence of carbon (graphite) fiber-related "accidents" and consequent "incidents" [Intergovernmental Committee, 1978, p. 26] are not in themselves aerospace applications of composites, but nonetheless may be attributable to such activities. The incidents are a direct consequence of the high electrical conductivity of carbon fibers, and are included here for their significance. The incidents are reported verbatim from "Carbon Fiber Study," NASA TM78718, pages 17 and 18. Said report offers an excellent summary of state-of-the-art knowledge and research on potential hazards and improvements.

The fiber hazard study has received and continues to receive significant attention from many agencies, and follows:

The discovery of the damaging effects of the impingement of carbon fibers on electrical equipment came about quite accidentally. In 1968, the Air Force was conducting electronic interference experiments. Test personnel noted that electronic equipment exposed to the carbon fibers malfunctioned. Laboratory tests substantiated the effects of carbon fibers on electrical equipment.

On May 12, 1972, during a clean-up of an area in the Union Carbide's Carbon Fiber Production building at Fostoria, Ohio, a cardboard carton containing untwisted filaments of fine strands or carbon fibers 6 to 42 inches long was inadvertently placed in the plant incinerator rather than going to land fill. Subsequently, fibers were emitted from the stack, and conveyed by air over the plant and surrounding areas. The loose fibers were experimental material from development work; carbon fiber composite material was not involved. The electrically conductive strands settled on several electrical substations at the plant, causing short-circuits and power outages in three of the substations. The carton of fibers was placed on a burning pile of wood in the incinerator with a stack about 125 feet high and five feet in diameter at the top. The incineration occurred at 9:00-9:15 a.m., on May 12, 1972, a

Friday. Within minutes, fiber strands were being discharged from the stack top. Strands (both single and in tumble-weed like clumps) were observed floating around in the air, settling on wires, roofs, and in the yard area. Some unburnt fibers were pulled from the fire and the fire was extinguished. The fibers were probably in the incinerator for 20 to 40 minutes. Between 9:25 a.m. and 10:00 a.m., the 4000-volt circuits in two substations at the Union Carbide plant about a half mile downwind from the stack were shorted out. Power was restored in these substations by 2:00 p.m. A third substation nearby shorted out at 5:15 p.m. on the same afternoon. Startup of this substation was delayed until the following day pending an inspection of temporary repairs. Eight other open air substations and a metering station at the plant which were downwind from the stack did not short out. A newspaper account in the Fostoria Review Times on May 12, 1972, reported the Union Carbide power failure and also quoted a spokesman for the Ohio Power Company as indicating that three or four power outages were reported in the city at 9:10 a.m. The Ohio Power Company spokesman also said failures occurred within 15 minutes of each other at Ohio Power's east end substation, which is adjacent to the Union Carbide plant, and its west end substation which is some 3 miles to the west (downwind) of the plant. A power outage was also reported on Saturday, May 13, 1972, from 1:25 p.m. to 3:17 p.m. which darkened most of the downtown area of the city east of Main Street. Ohio Power's east end substation adjacent to the Union Carbide plant was reported as losing six 4000-volt line insulators.

The impact of such incidents upon the state-of-the-art results from the extensive hazard study which ensued and the subsequent actions taken in response to the potential hazard. NASA limited further placement of new carbon fiber structural parts on aircraft until further information could be gained from the hazard study.

Current research activities by NASA, DOD, and others are aimed at more detailed assessment of the hazard in aerospace, surface transportation, sporting goods, and other industries. Methods of protection, and alternative or improved material properties are also being researched, including:

1. Develop insulating coatings for fibers.
2. Develop resin which chars and hangs onto fibers rather than releasing them.
3. Develop other fibers.

4. Mix in low  $T_{melt}$  fibers of glass to make a sticky glob.
5. Enclose the laminate in containing materials.
6. Other

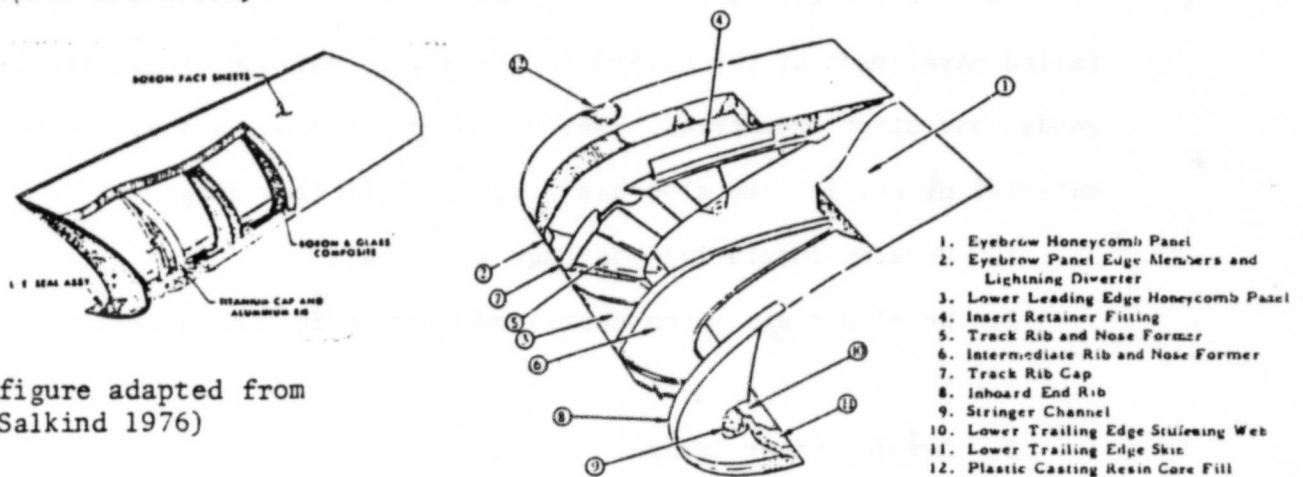
(Intergovernmental Committee 1978, Hardrath p. 4,5 in Bergamo Center 1978, NASA TM78652, Time Magazine 13 March 1973 p. 90, Landa 1978).

Cessna Miscellaneous Parts, apparently several parts are used:

Cessna 310 Radomes, kevlar with nomex core  
Cessna 404 wingtips, kevlar with nomex core  
Cessna 500 seats  
Cessna 650 (Citation III) plans for flaps, nacelles, Radomes, doors, wingtips,

Reported advantages include weight, transmissivity, cost, lightning protection, stiffness (Shearer 1977, Furnish 1977).

C-5A (Galaxy) Wing Leading Edge Slat, Lockheed-Marrietta, USAF, B/E skins on Al honeycomb core, Al ribs with titanium cap, some use of G1, 22% weight savings, erosion of resin on leading edge has occurred, moisture in some cases entered and corroded the core, some delaminations of skin from core occurred, maintenance errors have caused incidences of damage, foreign object damage (probably bird strikes) has occurred (Salkind 1976, Hackman 1973, personal communication with R. A. Pride, Advanced Composites Design Guide, Third Edition 1973 Vol. V, Section 5.1.2.2.3).



SUBSCALE MODEL BORON SLAT

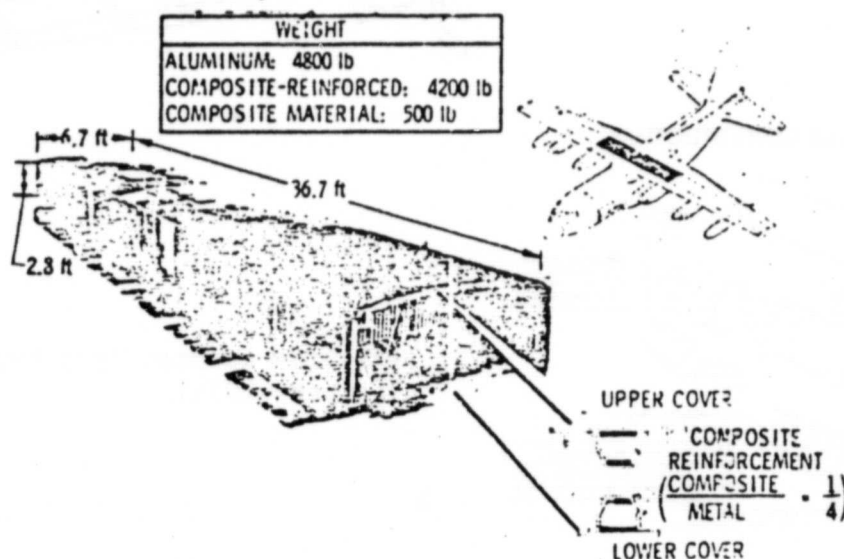
(figure adapted from  
Salkind 1976)

(figure adapted from USAF)

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C-130 (Hercules) Transport Center Wing Box Reinforcement, Lockheed-Georgia

Company, B/E used as selective reinforcement, box is  $11 \times 2 \times 0.9$  m ( $36.7 \times 6.7 \times 2.8$  ft), reinforcing strips applied to full length of both covers and on the hat section stringers, 2018 kg (4440 lb) 8% composite, 10% weight saved, 3 units built, one fatigue tested to 4 lifetimes, then fatigue tested 1 lifetime with artificially introduced cracks and delaminations, then residual strength tested, in addition to some smaller component tests, qualifying the other two boxes for in-service use starting in October 1974 with USAF 314th TAW and USAF Little Rock AFB, inspected at 6 month intervals for 6 years, no service damage or failure has been found (Dexter, "Review of NASA Composite Flight Service and Aircraft Energy Efficiency Programs," from Hardrath and Dexter 1977, Pride 1978, Hardrath 1977, Salkind 1976, Advanced Composites Design Guide, Third Edition 1973 Vol. V, Section 5.1.2.2.6).



(figure adapted from USAF)

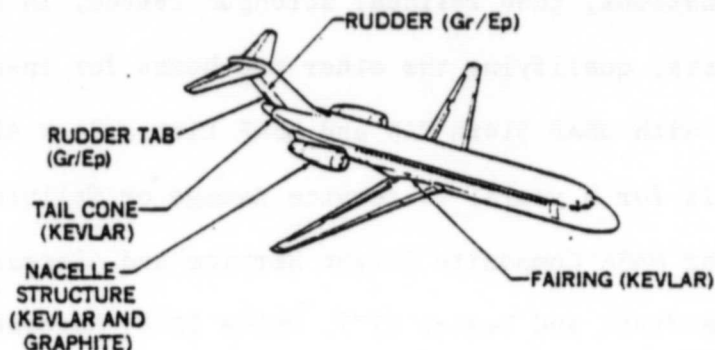
C-141 (Starlifter) Landing Gear Top Doors, B/E honeycomb core, about  $.6 \times 1.2$  m ( $2 \times 4$  ft) (personal communication with R. A. Pride).

DC-9 Miscellaneous Parts, McDonnell-Douglas, several parts are developed and/or in service:

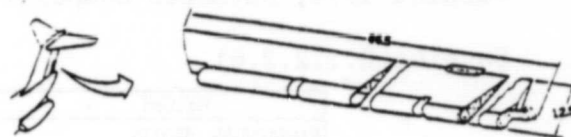
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tail cone, K/E skins on honeycomb core, 9 kg (20 lb) lighter  
wing fairing, K/E, 27 kg (60 lb) lighter  
nacelle structures including nose cowl outer barrel (Rohr Industries)  
upper and lower cowl doors, and apron assembly, all use K/E skins  
with sandwich beam stiffeners  
rudder, Gr/E, 49 kg (108 lb) lighter  
rudder tab, Gr/E, 2.7 kg (6 lb) lighter, 2.2×.3 m (86.5×12.5 inches)  
in development

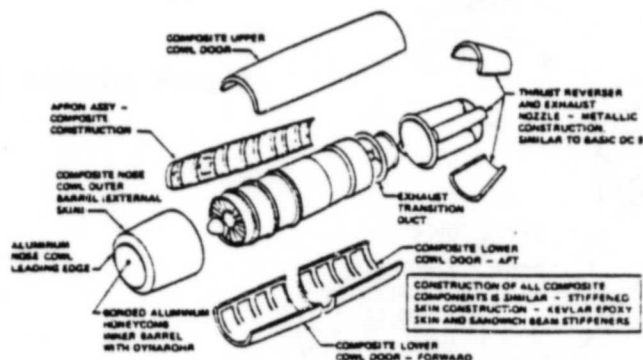
(Douglas Aircraft Company 197x).



DC-9 Rudder Tab



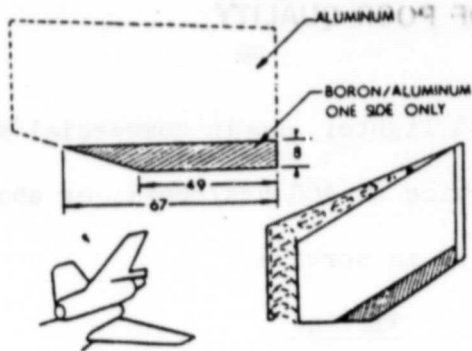
DC-9-80 NACELLE CONSTRUCTION



(figures adapted from Douglas Aircraft Company 197x)

DC-10 Aft Pylon Skin, McDonnell-Douglas Aircraft Company 3B/Al skin panels have been installed on United Airlines just above the center engine, exposed to elevated temperatures and high acoustic loadings, 170×20 cm (67×8 in), 1.6 kg (3.5 lb), 11 plies, 0, ± 45, 90, 26% lighter, satisfactory operation since service began in August 1975 (Elliot 1977, Douglas Aircraft Company April 1978, Hardrath 1977).

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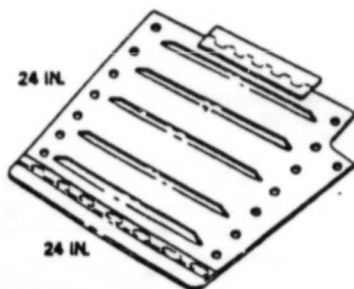


(figure adapted from Elliot 1977)

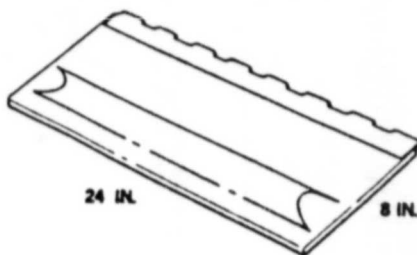
DC-10 Miscellaneous Parts, McDonnell-Douglass,

Wing Trailing Edge Panel, .6×.6 m (24×24 in), testing  
Vertical Stabilizer Trailing edge panel, .6×.2 m (24×8 in), 5 panels  
in service with no problems  
Nose Landing Gear Door, 1.2×.6 m (48×24 in), being developed  
Floor Beams and Struts, Gr/E, tests completed, one installed in  
Balair ship no. 267 July 15, 1978

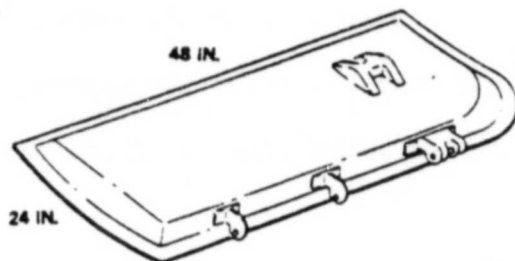
(Douglas Aircraft Company 197x).



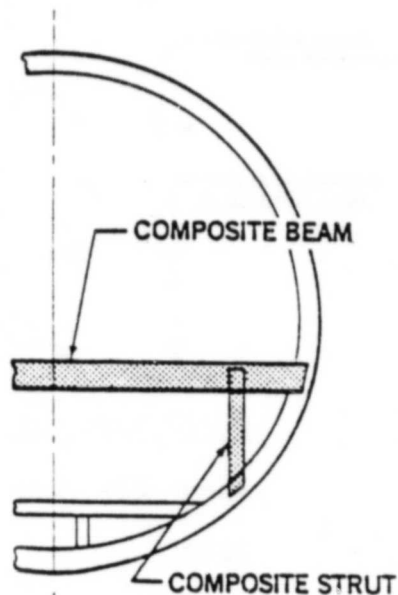
WING TRAILING EDGE  
PANEL



VERTICAL STABILIZER  
TRAILING EDGE PANEL



NOSE LANDING GEAR DOOR



(figures adapted from Douglas Aircraft Company 197x)

DC-10 Upper Aft Rudder, McDonnell-Douglass, 4.1×1 m (13.5×3 ft), Gr/E skins with  
multiple ribs and 2 spars, substructure, G1 leading and trailing edges, mechanically

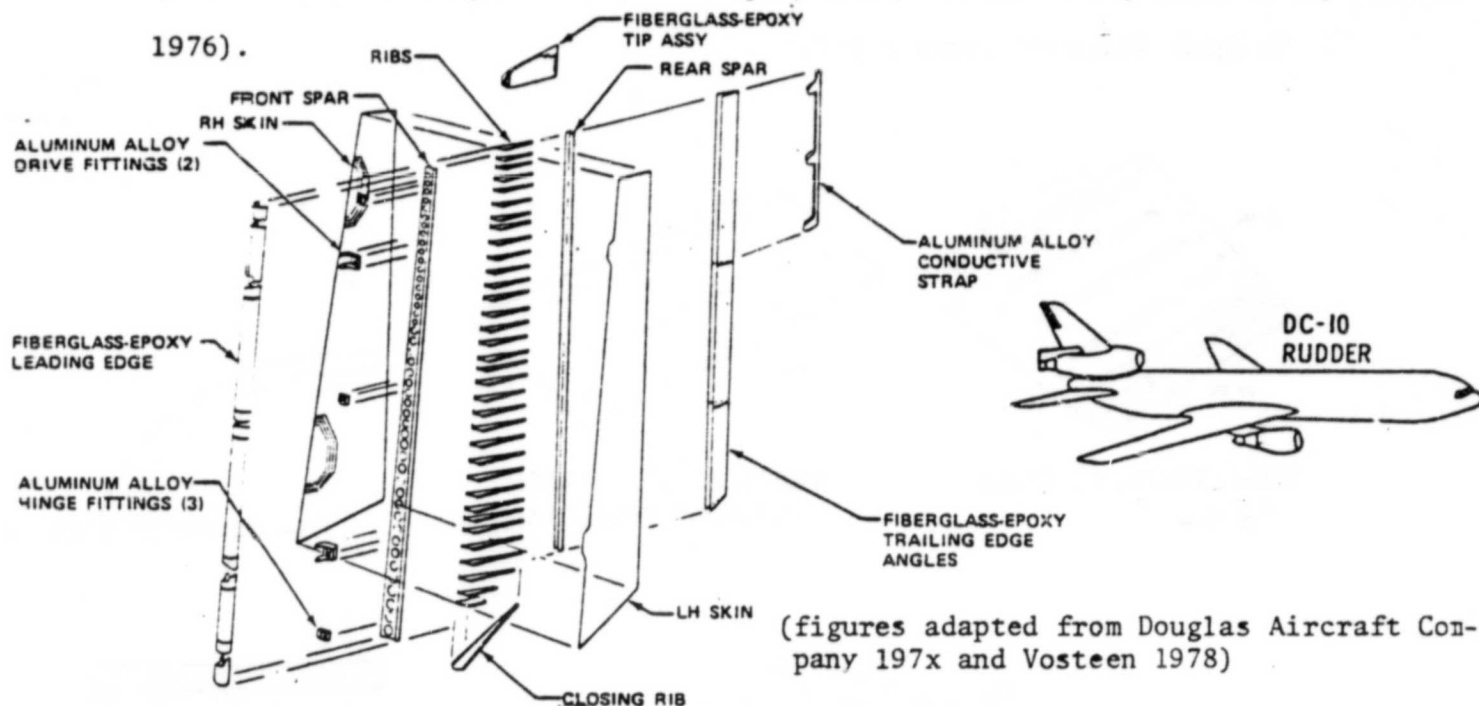
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fastened, aluminum fittings, at least 30% lighter, one in commercial service since 1976, one has been acquiring flight service at 400 hrs/month, or about 13 hrs/day, 26 kg (57 lb), 77% composite, 9 more are in service

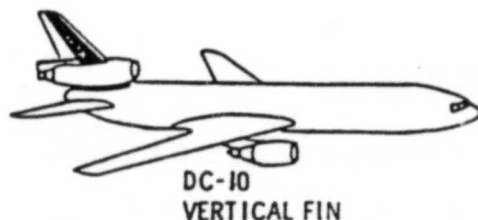
<u>Aircraft</u>	<u>Airline</u>
1	Korean
4	New Zealand
1	Swiss Air
3	Trans International
1	Western

(Hardrath 1977, Douglas Aircraft Company 197x, Vosteen 1978, Pride 1978, Salkind 1976).



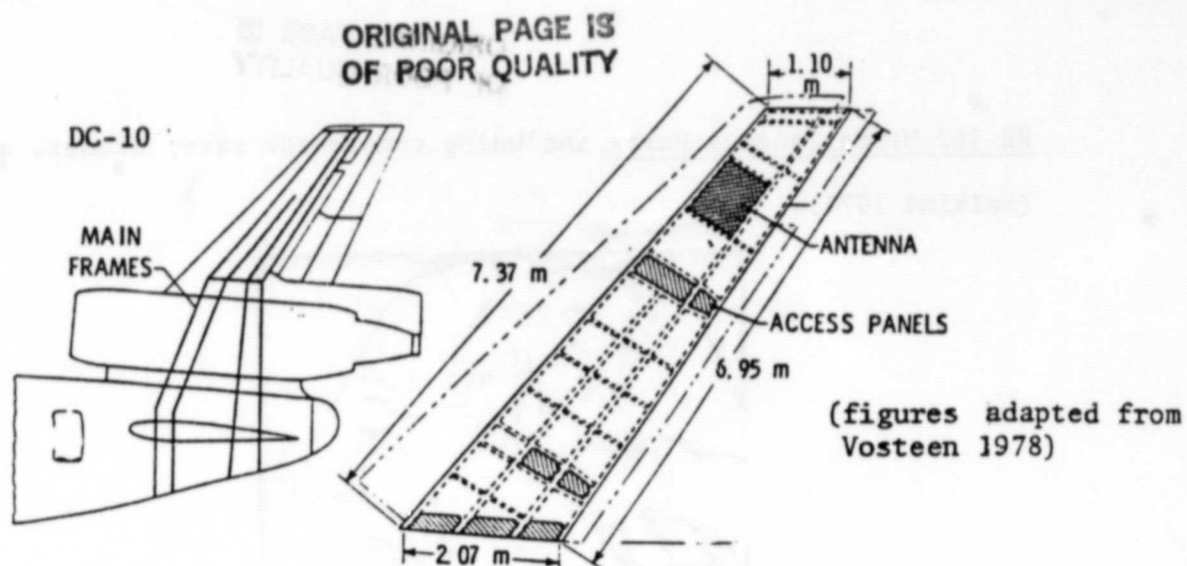
(figures adapted from Douglas Aircraft Company 197x and Vosteen 1978)

DC-10 Vertical Stabilizer, McDonnell-Douglas Aircraft Company, being developed, about 27% lighter, particular attention being paid to joints with metal (Vosteen 1978).



(figure adapted from Vosteen 1978)





D-36 (Darmstadt also ASW-12) Sailplane, early 1960's German Gl application (Hodges 1977).



(figure adapted from Hodges 1977)

#### Engines Miscellaneous Parts

Miscellaneous attempts to apply composites to primary and secondary aircraft turbine engines structure have been made. Incentives include potential weight reduction, centrifugal stress reduction, and promises of improved fatigue life.

Numerous applications of both metal matrix and organic matrix composites to rotating blades, vanes or static blades, and various stator parts have been considered (Advanced Composites Design Guide, Third Edition 1973 Vol. V, Section 5.4) (Renton 1977). One major impedance to the successful application of composites to blades has been inadequate impact strength, particularly soft body impact. Much current research concerns improved impact or "FOD" properties.

The following is a sampling of aerospace development efforts for composites for engines (more are reported in the references):

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RB-162 Miscellaneous Parts, including compressor case, blades, and vanes (Salkind 1973).



*Compressor rotor of RB 162 engine with fiberglass composite blades (Rolls-Royce, Ltd.).*

(figure adapted from Salkind 1973)

RB-211 Turbine Engine Fan Blades, primary, Rolls-Royce. Rotating composite blades were not able to survive ingestion of all anticipated foreign objects. Composite bladed engines were to be used in L1011. The end result was much difficulty (personal communication with Dr. T. Cruse, Salkind 1976, Advanced Composites Design Guide Third Edition 1973, Section 5.4.2.1.5).

Pratt and Whitney JT9D-59 and JT9D-70 Engines Fan Exit Guide Vanes, Gr/E guide vanes 60% lighter than aluminum vanes reported to be first production commitment for composites in gas turbine engines, to power B-747's and DC-10's (Salkind 1976, Advanced Composites Design Guide Third Edition 1973, Section 5.4.2.1.4).

Pratt and Whitney F100 Engine, (used in F15 and F16) considering B/Al vanes (personal communication with Tobey Cordell).

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GE F103, GE and NASA developing hybrid SG1/Gr/E blades (personal communication with Tobey Cordell).

GE A10 Forward Stator, GE and Rohr have designed and built Gr/E, G1, and K forward stator (personal communication with Tobey Cordell).

Soviet MIG Jet Engines, reportedly use composite blades.

Fairchild Surveillance Drone, composite structure developed as integral fuel tank (Hackman 1973).

FS-28 (Avispa) Single Engine Pusher Miscellaneous Parts, University of Stuttgart in West Germany student project, featured composite landing gear and propeller, flew in 1972 and still flying (Hodges 1977).

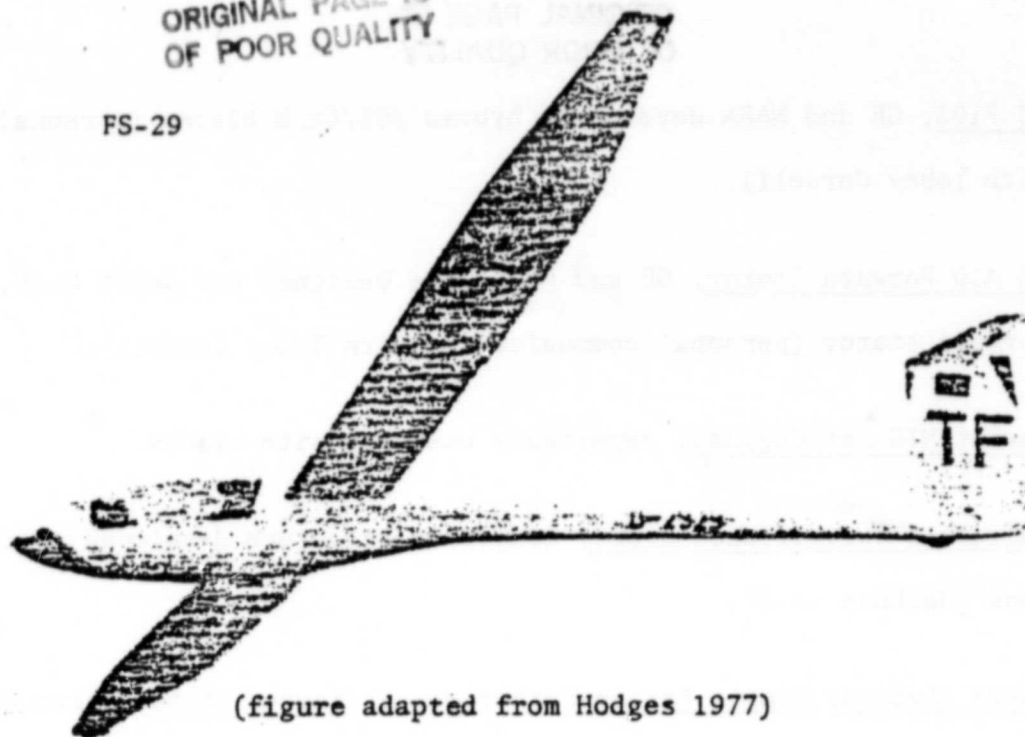


(figure adapted from Hodges 1977)

FS-29 Sailplane Wing, primary, University of Stuttgart in West Germany, Gr wings design with variable span of 13 to 19 m (43 to 62 ft) required very small deflections to allow outer panels to slide over inner panels, flew in June 1975 and still flying in university flying club (Hodges 1977).

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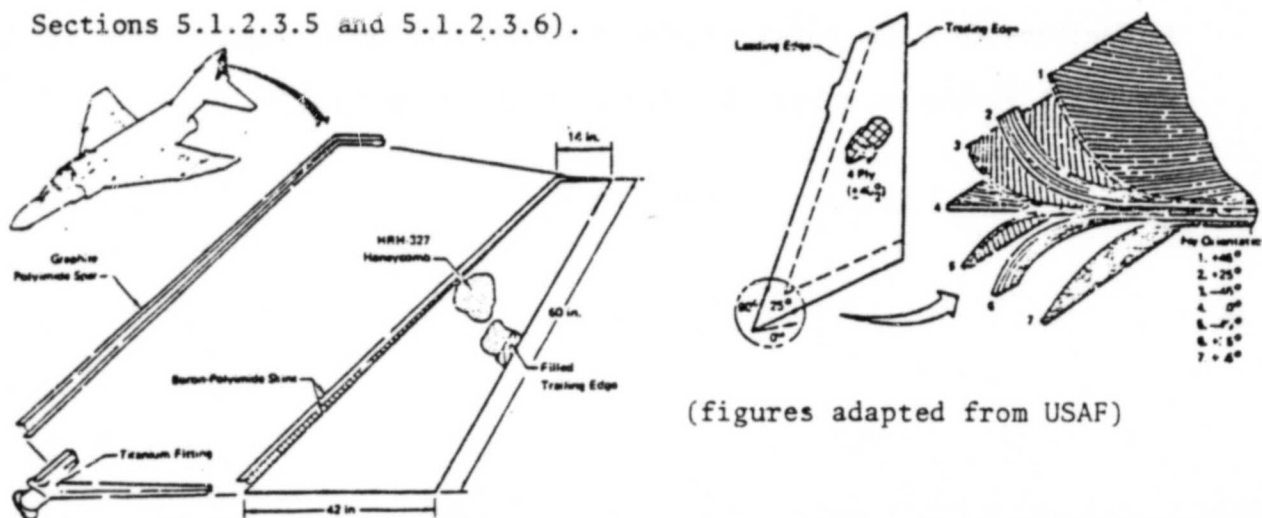
FS-29



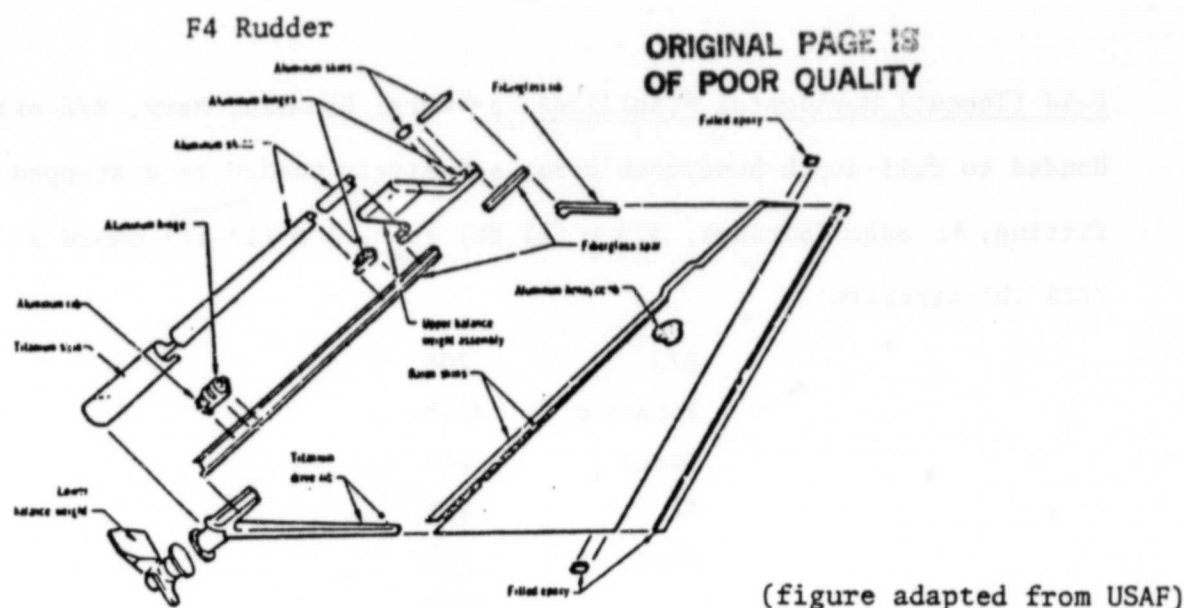
(figure adapted from Hodges 1977)

F4 (Phantom) Fuel Access Doors, McDonnell-Douglass, Navy, Gr/E with various metal fasteners, adhesives, honeycomb core, and moisture content specimen break-off tabs to test corrosion prevention methods where Gr/E is in proximity to Al, 4 sets of doors about 1' diameter have been in service on carrier-based aircraft for 2 years and will be tested early 1979 (personal communication with Dr. M. S. Rosenfeld

F4 (Phantom) Rudder, McDonnell-Douglass, 4 ply B/E skins adhesively bonded to full-depth Al honeycomb core, 19 kg (41 lb), 37% lighter, 45 manufactured and placed in service beginning 1969, in acoustic fatigue tests outperformed metal rudder, Gr/B/polyimide version was also considered (Siegal 1977, personal communication with Frank Fechek, Advanced Composites Design Guide, Third Edition 1973 Vol. V, Sections 5.1.2.3.5 and 5.1.2.3.6).

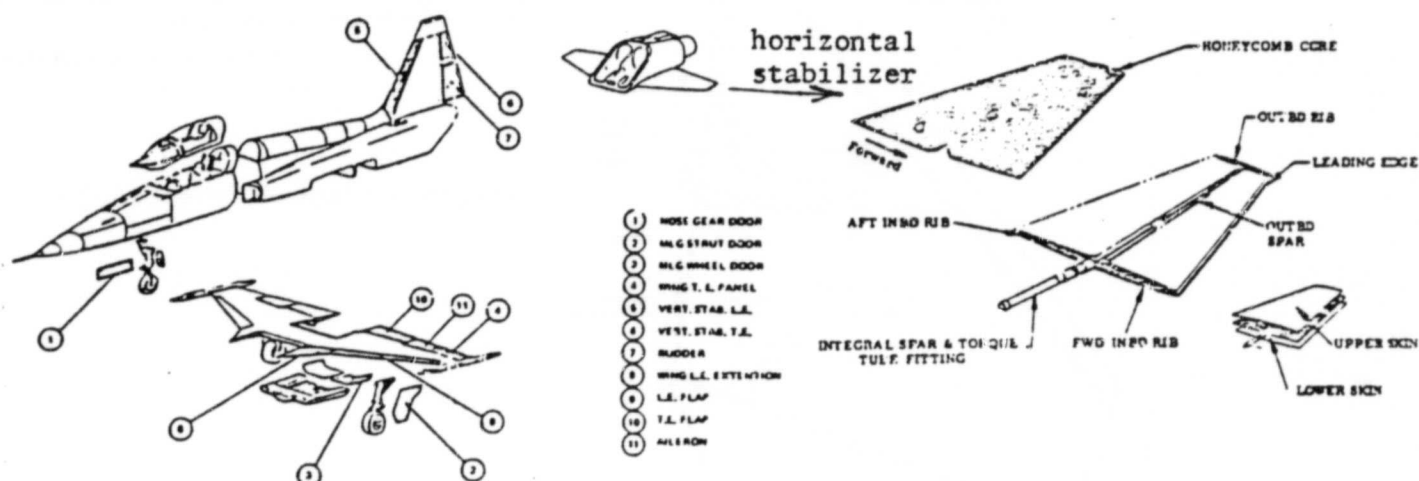


(figures adapted from USAF)



F-5 (Tiger) Aft Fuselage Mockup, General Dynamics, primarily Gr/E structure in which various concepts were studied including joint design, was a post-buckling design, was static tested, not flown (Salkind 1976, personal communication with GD).

F-5E Demonstration Components, Northrop Corporation, trailing edge flap was selected to demonstrate low cost manufacturability, studies included main landing gear strut door, speed brake, wing leading and trailing edge flaps, horizontal stabilizer, vertical stabilizer main spar (Advanced Composites Design Guide, Third Edition 1973 Vol. V, Sections 5.1.2.1 and 5.1.2.3.4).

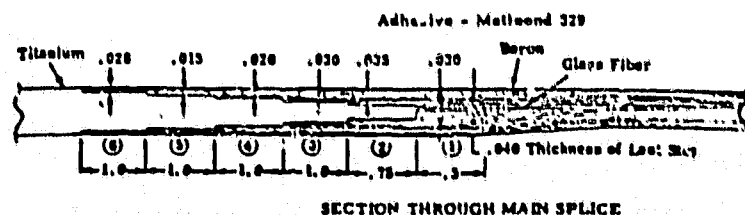
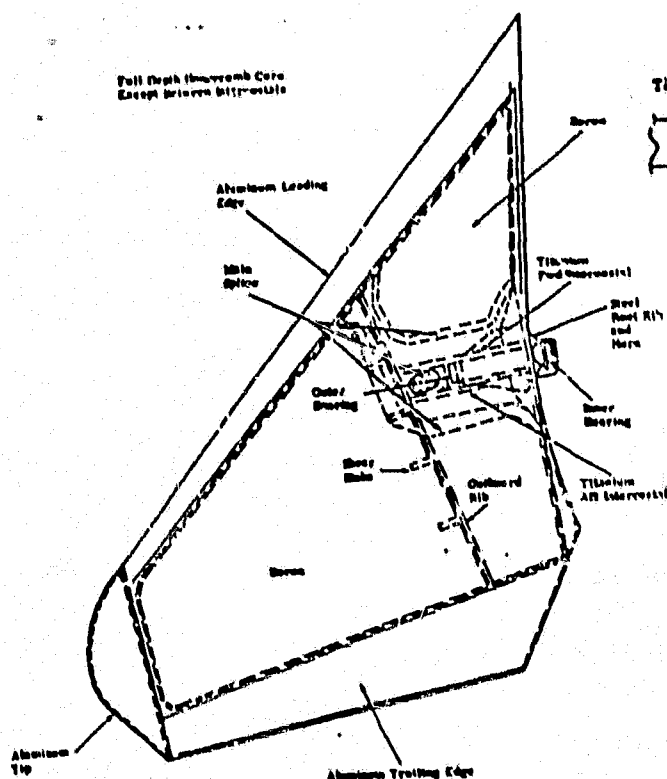


(figure adapted from USAF)

F-14 (Tomcat) Horizontal Stabilizer, primary, Grumman, Navy, B/E skins adhesively bonded to full-depth honeycomb core, adhesively bonded to a stepped metal pivot fitting, Al edge fairings, 3.3 m (11 ft) span, 4 m (13 ft) chord at root, 375 kg (825 lb)/aircraft:

B/E	20%
Titanium	28%
Steel	14%
Al	24%
Misc.	14%
	100%

18% lighter than metal, 21 Dec., 1970 first flight, fleet introduction 14 October 1972, 1264 skins completed, 110,000 flight hours accumulated, most damage to LE, TE, and tip (dents, corrosion), no honeycomb core water problems reported, 1 delamination caused by debris impact repaired and returned to service, field reports on repair kit say they're excellent, some have received gouges from forklifts, some delamination of electromagnetic foil strips has occurred (Hadcock 1972, Corvelli 1977, Salkind 1976, personal communication with NADC personnel, Advanced Composites Design Guide, Third Edition 1973, Section 5.1.2.3.2).

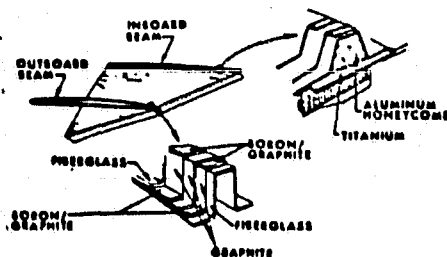


(figures adapted from USAF)

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F-14 (Tomcat) Main Landing Gear Door, secondary, Grumman, Navy, Gr/E skins on Al honeycomb core, nine ship sets will be installed and monitored periodically (Corvelli 1977, personal communication with NADC personnel).

F-14 (Tomcat) Overwing Fairing, secondary, Grumman, Navy, Gr/Gl/E skins with integrally molded B/E beam, on Al honeycomb core, Gl is placed over Gr for corrosion protection, 25% lighter and 40% cheaper, curved flexible Gr/E seals (fingers) about 10" long ride on moveable wing to seal out air, seals may be production items on metal planes, 5 ship sets of the fairings are scheduled to have been installed and flow by this publication, periodic monitoring is planned (Salkind 1976, personal communication with Dr. M. S. Rosenfeld).



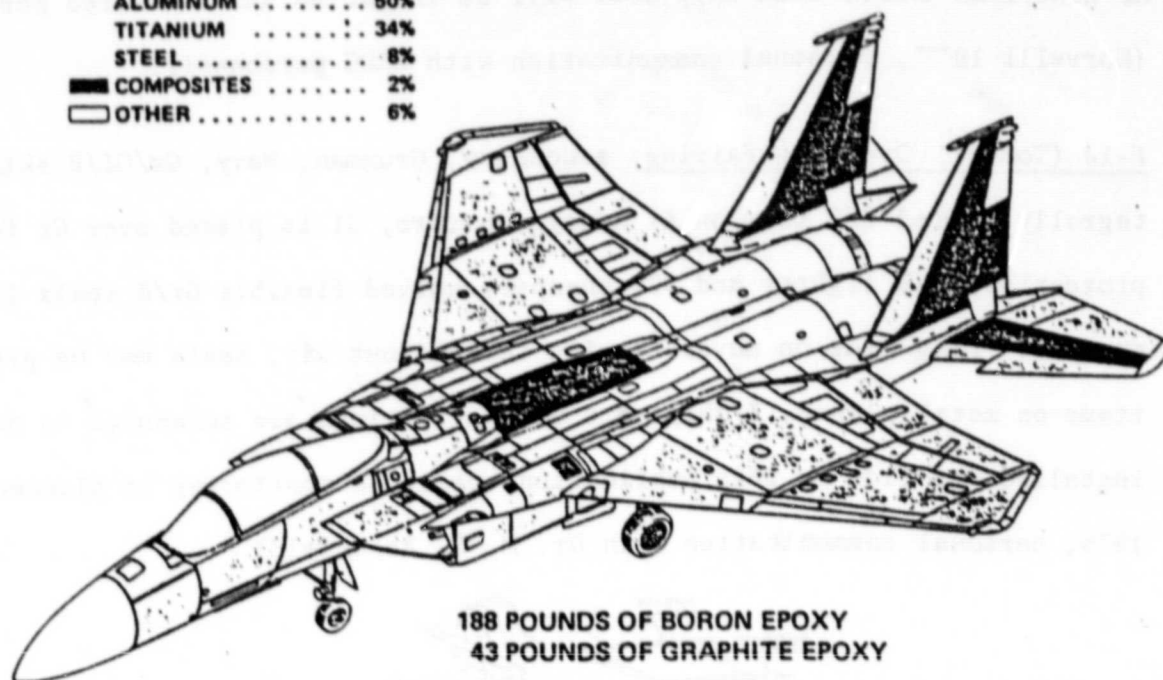
(figure adapted from Salkind 1976)

F-15 (Eagle) Horizontal Stabilizer, McDonnell-Douglas, USAF, B/E skins on Al honeycomb core, bonded root joint to titanium, approximately 2.4 m (8 ft) long x 3.6 m (12 ft) chord at root, some minor problems have been reported, 528 units (2 per aircraft) have been produced, 59 kg (130 lb) lighter than 339 kg (746 lb) metal component in service since 1972 (Siegal 1977, Douglas Aircraft Company 197x, Salkind 1976, personal communication with Frank Fechek, Hinkle 1978 from Bergamo Center 1978, pp. 152-158).

## F-15 MATERIAL DISTRIBUTION

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STRUCTURAL WEIGHT	
ALUMINUM .....	50%
TITANIUM .....	34%
STEEL .....	8%
COMPOSITES .....	2%
OTHER .....	6%



(figure adapted from Siegal 1979)

F-15 (Eagle) Rudders, McDonnell-Douglass, USAF, 548 units have been produced (2 per aircraft) and in service since 1972 (Siegal 1977).

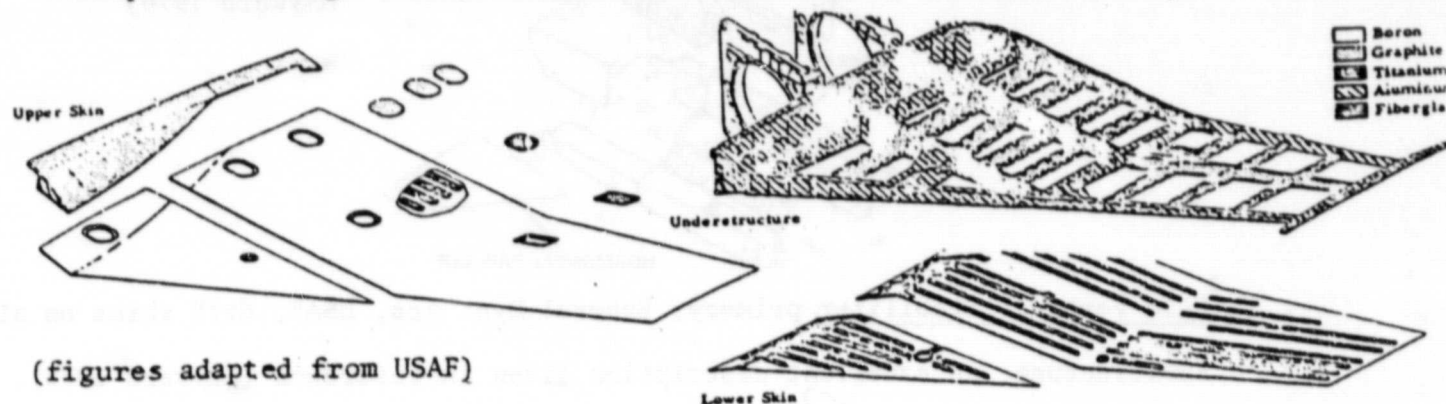
F-15 (Eagle) Speed Brake, secondary, McDonnell-Douglass, USAF, Gr/E skins on honeycomb core, 8.6 kg (19 lb) lighter than 51 kg (112 lb) metal part, less tools, less fasteners, less parts, 233 units have been produced, has been a production item since ship number 60, 9 repair incidents have occurred, maintenance handling has required some repairs, the speedbrake is a natural place to step when walking from one wing to other, 3 repairs have been documented (Siegal 1977, Douglas Aircraft Company 197x, personal communication with Frank Fechek).

F-15 (Eagle) Vertical Stabilizer, McDonnell-Douglass, USAF, B/E skins on Al honeycomb core, some minor problems have been reported, 542 units (2 per aircraft) have been produced, 61 kg (135 lb) lighter than 277 kg (610 lb) metal part, in service since 1972 (Siegal 1977, Salkind 1976, Douglas Aircraft Company 197x, personal communication with Frank Fechek).



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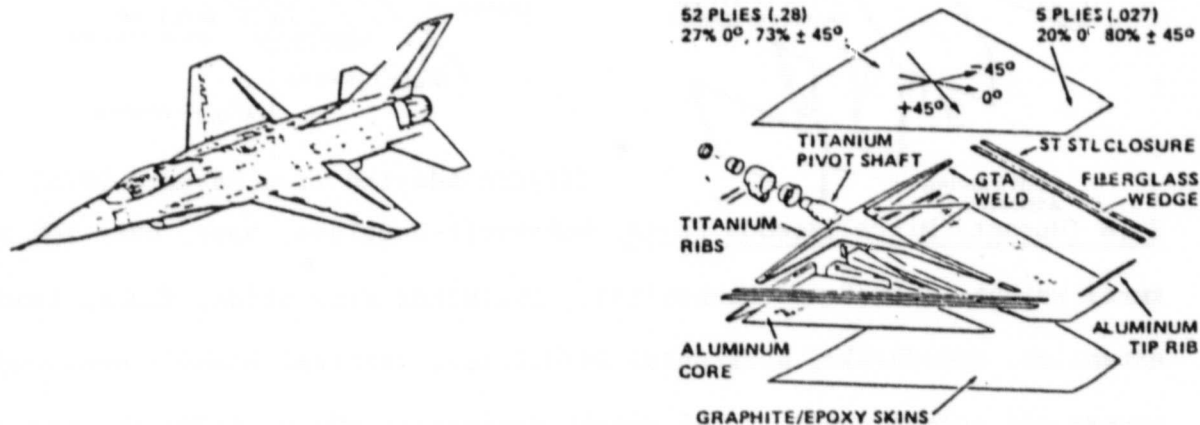
F-15 (Eagle) Wing, McDonnell-Douglas Corporation, USAF, design, manufacture, ground test, and flight test of B/E and Gr/E and metal alloy torque box (Advanced Composites Design Guide, Third Edition 1973 Vol. V, Section 5.1.2.2).



(figures adapted from USAF)

F-15 (Eagle) Wing- and Empennage-to-Fuselage Attachment Fittings Component Study, General Dynamics, AFML, feasibility study (Advanced Composites Design Guide, Third Edition 1973 Vol. V, Section 5.1.2.5.1).

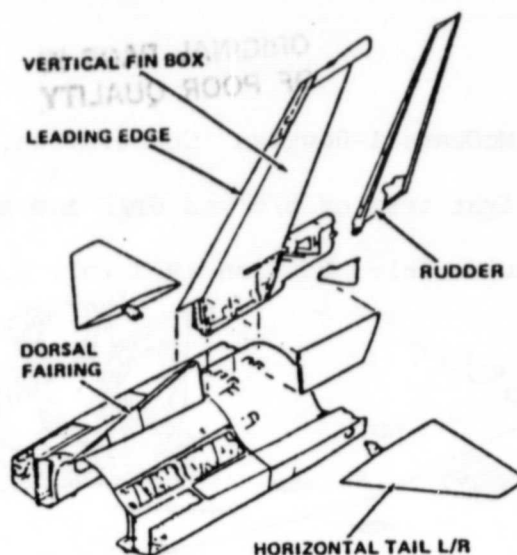
F-16 Horizontal Tail, primary, General Dynamics, USAF, L and R are interchangeable, see reference for details (Hayward 1976).



(figures adapted from Hayward 1976)

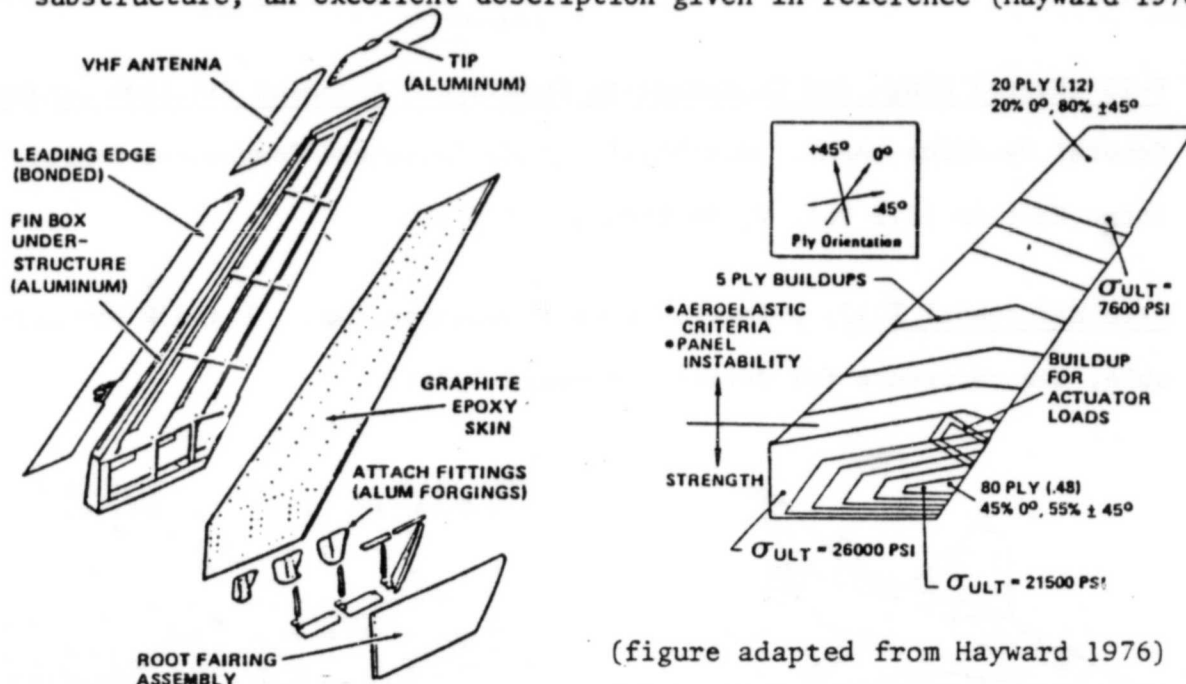
F-16 Rudder, General Dynamics, USAF (Hayward 1976).

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(figure adapted from  
Hayward 1976)

F-16 Vertical Stabilizer, primary, General Dynamics, USAF, Gr/E skins on al substructure, an excellent description given in reference (Hayward 1976).

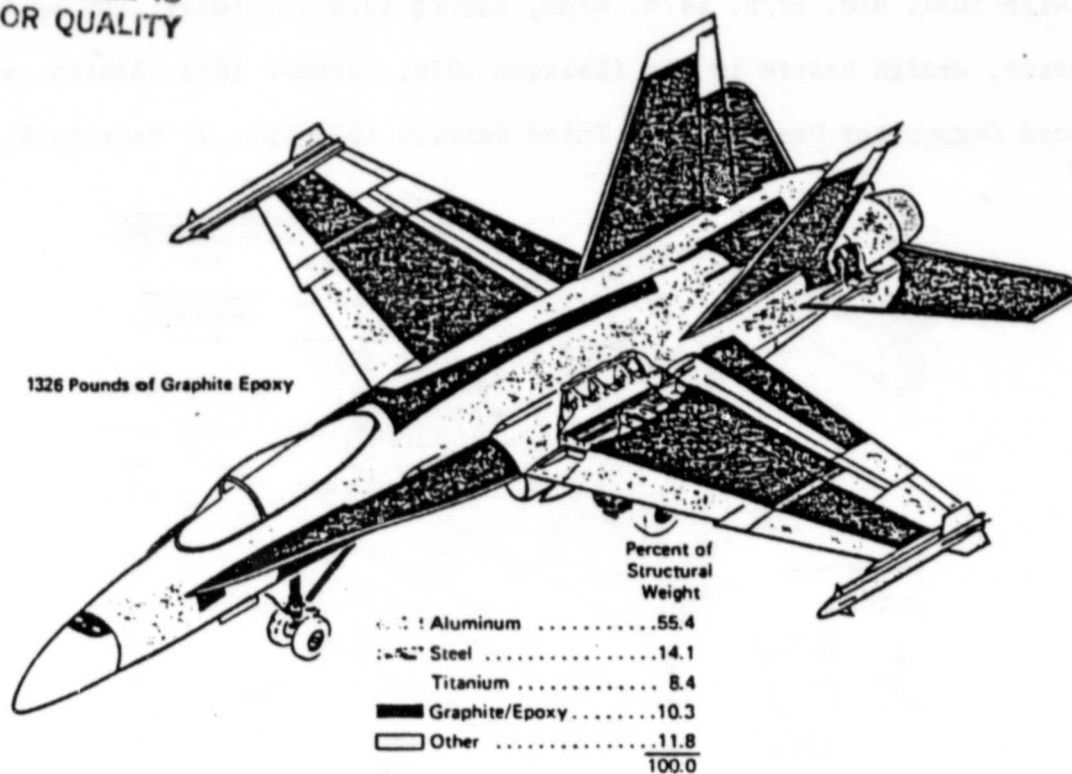


(figure adapted from Hayward 1976)

F-18 (Hornet) Miscellaneous Parts, McDonnell-Douglass, Navy, over 10% of structural weight is Gr/E and composites, including wing skins, flaps, leading edge extension, speedbrake, horizontal stabilizer, vertical stabilizers, rudders, covers and access doors, total weight savings is 800 kg (1761 lb) for equal performance to metals, part of which is direct and part of which is due to downsizing, most production items will be in service in about 3 years (Douglas Aircraft Company 197x, Siegal 1977).

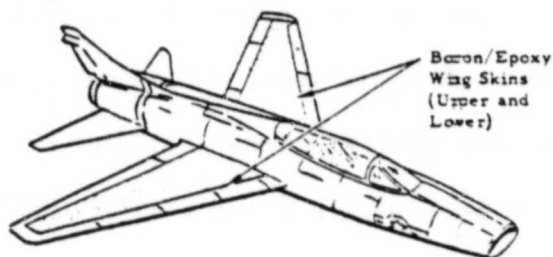
## F-18 MATERIALS DISTRIBUTION

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(figure adapted from Siegal 1977)

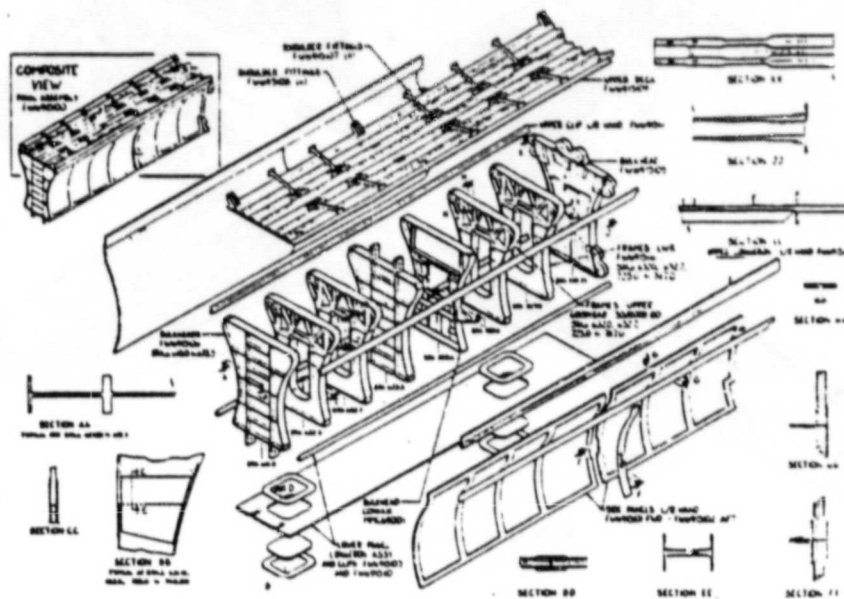
F-100 Wing Skin, primary, International Rockwell, B/E upper and lower skins developed to replace metal, 5.8 m (19 ft) long, 1.5 m (5 ft) chord at root, test program (Hackman 1973, Advanced Composites Design Guide, Third Edition 1973 Vol. V, Section 5.1.2.2.2).



(figure adapted from Hackman 1973)

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F-111 Aft Fuselage Mockup, General Dynamics, USAF, 4 m (160 inch) long, pressurized with fuel, B/E, Gr/E, G1/E, B/A1, 418 kg (920 lb) total, 209 kg (460 lb) composite, design tested in rig (Salkind 1976, Hackman 1973, Ashton, et al. 1972, Advanced Composites Design Guide, Third Edition 1973 Vol. V, Section 5.1.2.4).



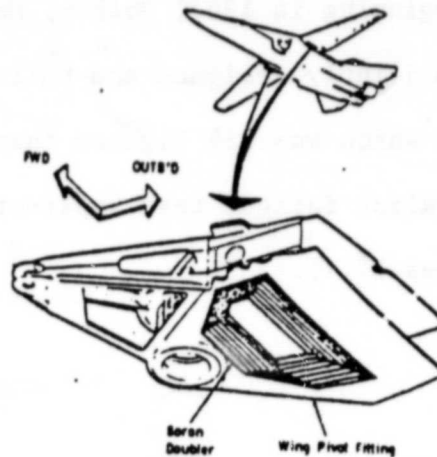
(figure adapted from Hackman 1973)

F-111A Horizontal Stabilizer, primary, General Dynamics, USAF, B/E skins bonded to full-depth honeycomb core, bonded and bolted to titanium pivot attachment fitting, flutter critical and therefore designed for both stiffness and strength, survived 4 lifetimes in fatigue test, 30% lighter, one ship set weighs 380 kg (837 lb), some water has entered cores (Salkind 1976, Hadcock 1972, Hackman 1973, Advanced Composites Design Guide, Third Edition 1973 Vol. V, Sections 5.1.2.3 and 5.1.2.37).

F-111B Wing Box Extension, Grumman Aerospace, USAF, B/E demonstration structure developed, was static and fatigue tested (Advanced Composites Design Guide, Third Edition 1973 Vol. V, Section 5.1.2.2.7).

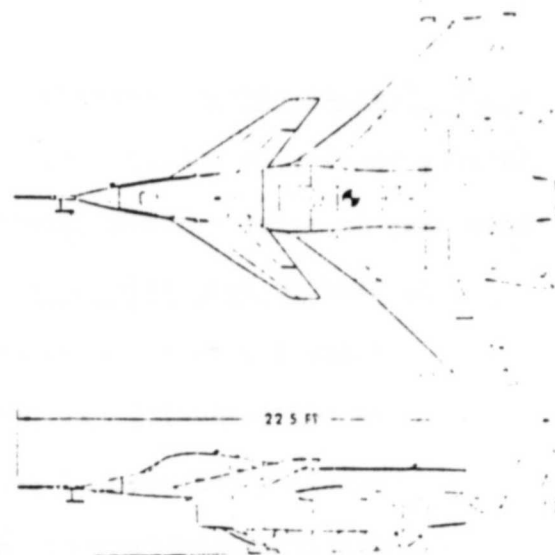
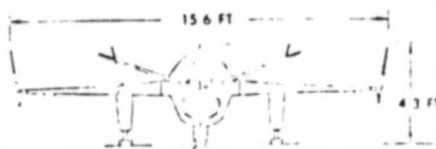
F-111 Outboard Wing Pivot Doubler, General Dynamics, USAF, B/E doubler was retrofitted to metal wing pivot fitting to decrease strain level and increase fatigue life, saved 11 pounds, saved 21% cost on production aircraft and 60% on retrofitted aircraft (Salkind 1976).

(figure adapted from  
Salkind 1976)



F-111 Wing Pivot Fairing, secondary, General Dynamics, USAF, Gr/E, 26% lighter, 31% less costly (Salkind 1976).

HiMat, primary, International Rockwell, two remotely piloted vehicles have been built and sent to NASA Dryden for tests, 35% of the structure is composites including aeroelastic tailored wings (Rockwell International 1977).

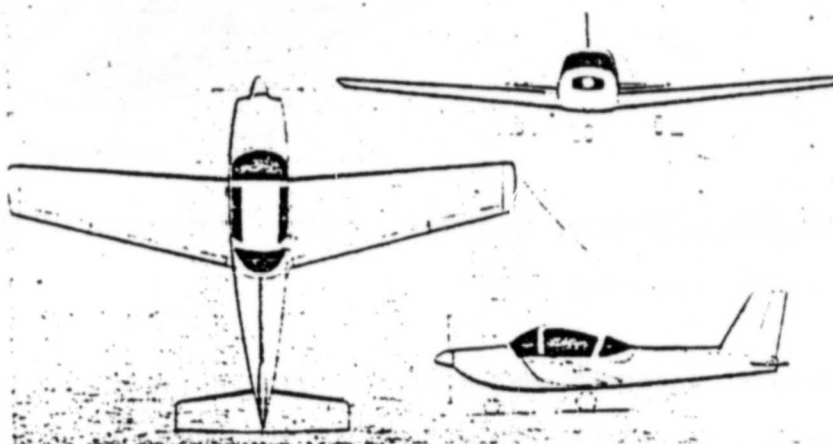


(figure adapted from Rockwell International 1977)

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Lear Fan, primary, developing an innovative mostly Gr/E airplane to be flown in May, 1980 (Loving 1979, personal communication with Dr. D. Konishi).

LFU-205, beginning in 1963, Bolkow, Messerschmitt, Blohm, Puter, and Rhein Flugzeugbau jointly designed and built an experimental 4-place partly composite light plane which was 15% lighter than metal, it flew in 1968, 36,000 flight hours equivalent fatigue test apparently caused no evidence of material deterioration (Hodges 1977).



(figure adapted from Hodges 1977)

L-1011 Fairing Panels, secondary, Lockheed California, NASA, three types of fairing panels, 18 panels total, have been installed on commercial aircraft and have been in service since January 1973:

- (a) Wing to Body Fairing Panel, K49/E skins on Nomex honeycomb core, sprayed with Al to discharge electrical potentials, 150×170 cm, (60×67 inches), 7 kg (16 lb) mass is 25% lighter than fiberglass panel it replaced, 2 per aircraft.
- (b) Center Engine Fairing Panel, K49/E skins on Nomex honeycomb core, higher temperature curing epoxy used because of proximity to engine, triangular 2 m×750 mm (30×82 inches), 2.3 kg (5 lb) mass is 30% lighter than fiberglass panel it replaces, 2 per aircraft.

- (c) Wing to Body Fillet, solid K49/E laminate, 2.3 mm thickness near middle tapered to .8 mm near edges, 830×140 mm (32.5×5.5 inches) 1 kg (2 lb) is 32% lighter than original part it replaces, 2 per aircraft.

In 5 years, these parts accumulated 12,138 flight hours with the following airlines

<u>Airline</u>	<u>Panels</u>	<u>Aircraft</u>
TWA	6	1
Eastern	6	1
Air Canada	6	1

One ship set was removed and reinstalled on a different aircraft due to a fire which damaged other parts of the original aircraft. No significant incidents or damage have been reported to date, and these parts have required less maintenance than standard parts (Salkind 1976, Pride 1978, Hardrath 1977, Advanced Composites Design Guide, Third Edition 1973 Vol. V, Section 5.1.2.4.3, Stone 1978).

Wing to Body Fairing Panel



Center Engine Fairing Panel



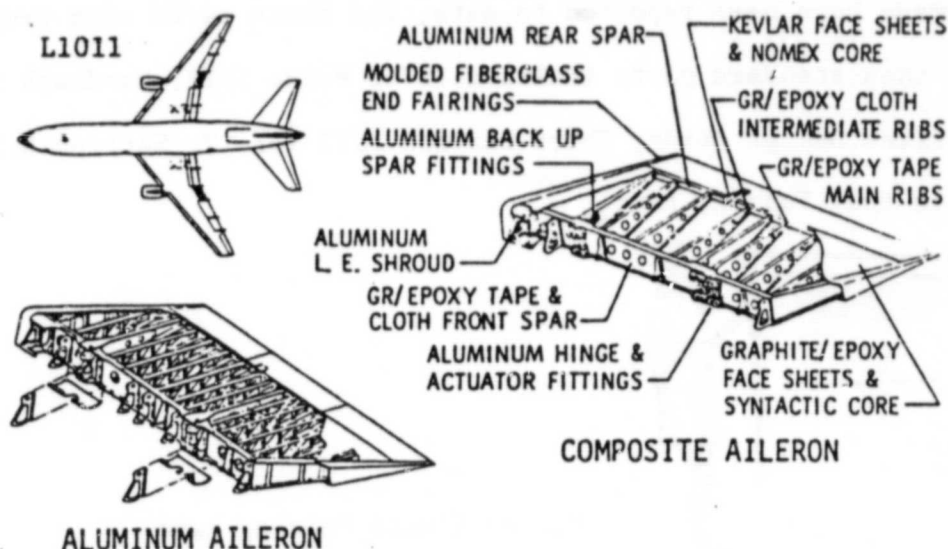
Wing to Body Fillet



(figures adapted from Hardrath 1977)



L-1011 Inboard Aileron, secondary, Lockheed California, NASA, to be located aft of the wing-mounted engine, one design has been selected for further development, Gr/E skins with understructure of combined Gr/E and aluminum ribs and spar pieces, filled with syntactic foam, preliminary tests show that foam core gives better impact strength than honeycomb core; no detectable damage at impact energies well above that produced by 1.8 cm diameter hailstone falling at terminal velocity, impact damage environment reportedly comes from hailstone impact on ground, preliminary works shows 25% potential weight savings and less parts and fasteners, 10 ship sets will be fabricated to estimate production costs (Vosteen 1978).



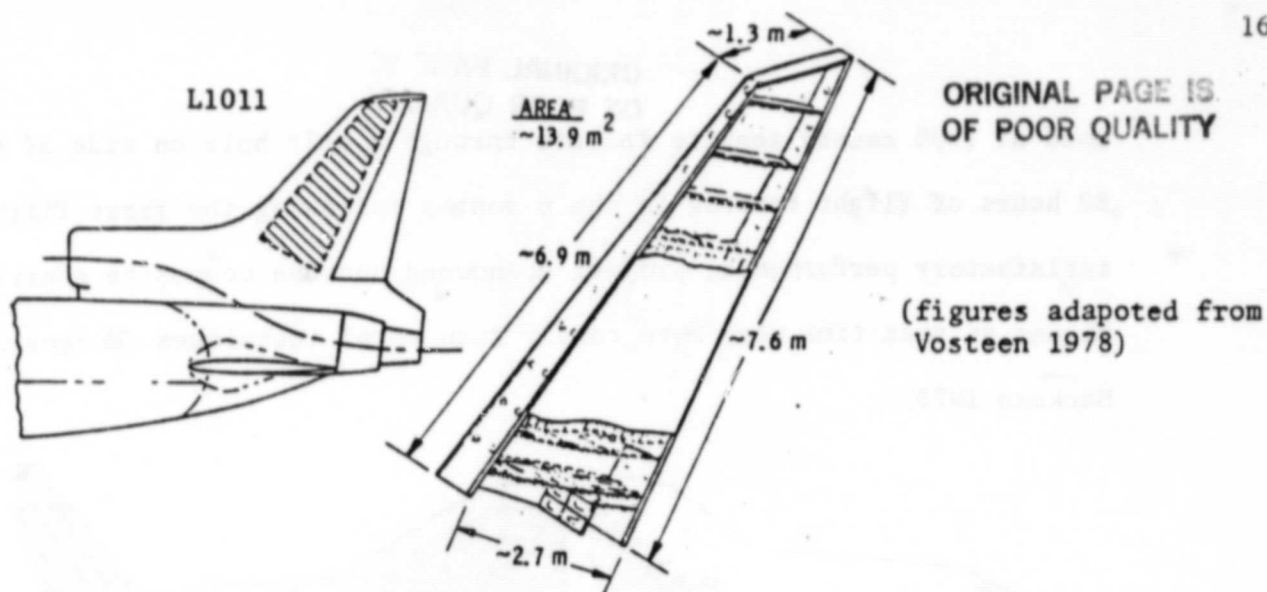
(figures adapted from Vosteen 1978)

L-1011 Vertical Stabilizer, primary, Lockheed California, being developed using Gr/E hat-stiffened skins, twin spar substructure with ribs, expected to be 28% lighter (Vosteen 1978, English 1977, 1978).



(figure adapted from Vosteen 1978)





Missiles, Miscellaneous Parts, development of various structural parts of missiles including bulkheads, casings, and interstages began in the 1960's and often resulted in significant weight savings, materials used included Gr/E, B/Al, and C/C, missiles having composite structure include Atlas, Polaris, Minuteman, MX, and Trident, more applications and details are given in the references (Salkind 1976, Hackman 1973, Petersen and Yarborough 1977, Laramee and Canfield 1972, Advanced Composites Design Guide, Third Edition 1973 Vol. V, Section 5.3).

OV-10A (Bronco) Wing Center Section, test section built for development of joints, full scale 2m (7 ft) long, represents fuselage to wing attachments, upper skin fuel cell door, scarf joint splice in lower skin, front and rear spar access holes reinforced with titanium rings, skins attached to spars using adhesive bond tongue and groove joint, 40% lighter (Hackman 1973).

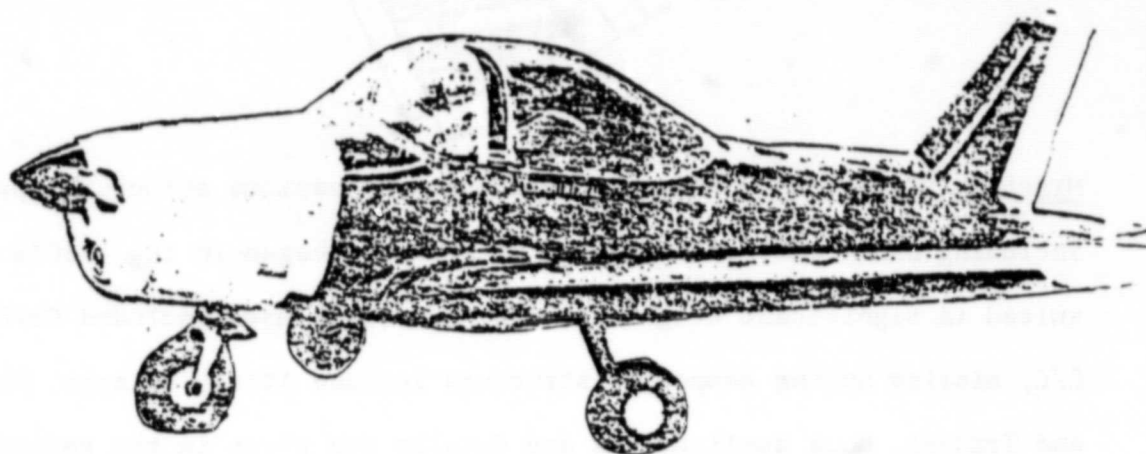


(figure adapted from Hackman 1973)

PA-29 (Papoose), primary, Piper Aircraft's Vero Beach Development Center, first flight 30 April 1962, following development started in 1958 to investigate processes and materials which might replace aluminum, G1/PE skins on paper honeycomb core wherever sandwich structure was used, wing static test failure occurred at 210% positive limit load, fuselage side load of 200% sustained, fuselage vertical

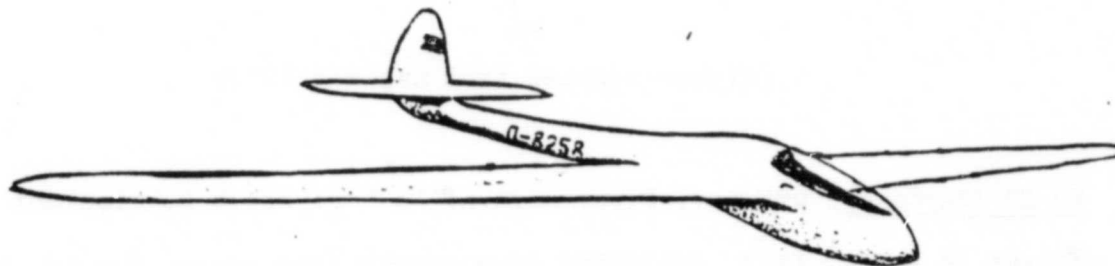
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load of 180% caused tensile failure through a bolt hole on side of the cockpit, 80 hours of flight testing in the 6 months following the first flight showed satisfactory performance, project abandoned because composite construction techniques at that time were more costly than metal techniques (Hodges 1977, Hackman 1973).



(figure adapted from Hodges 1977)

Phoenix Sailplane Wing, (primary), West German, reportedly first light aircraft to fly using composites in primary structure, first flight November 23, 1957, G1/PE skins on balsa wood core, proof tested at 4g's for 30 minutes and 1.3g's for 3 days without permanent deformation, airplane empty weight 164 kg (360 lb), evolved into Phoebus, total of 228 were produced up until 1970 at Bolkow (Hodges 1977).



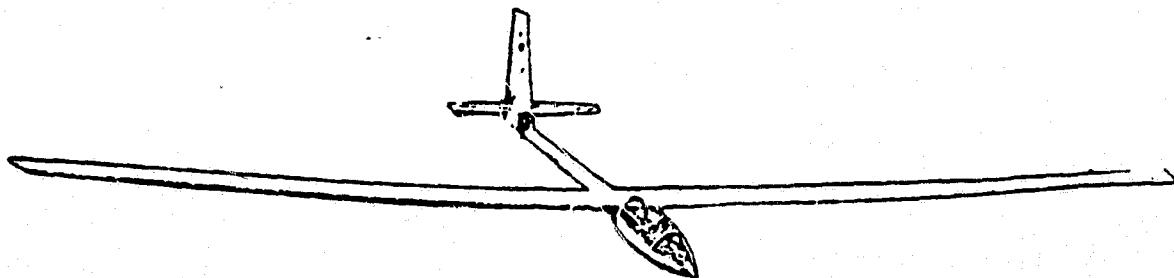
(figure adapted from Hodges 1977)

PiK-20D Sailplane Wing Spar Caps, Gr/E (Hodges 1977).



(figure adapted from Hodges 1977)

SB-10 Sailplane Wing, Akaflieg Braunschweig, West Germany, university students built, flew in 1972 and still flying, 29 m (95 ft) wing span, Gr/E 8 m center section, Gr/E torque box and spar caps, remaining 21 m (69 ft) was standard fiberglass sailplane construction (Hodges 1977).

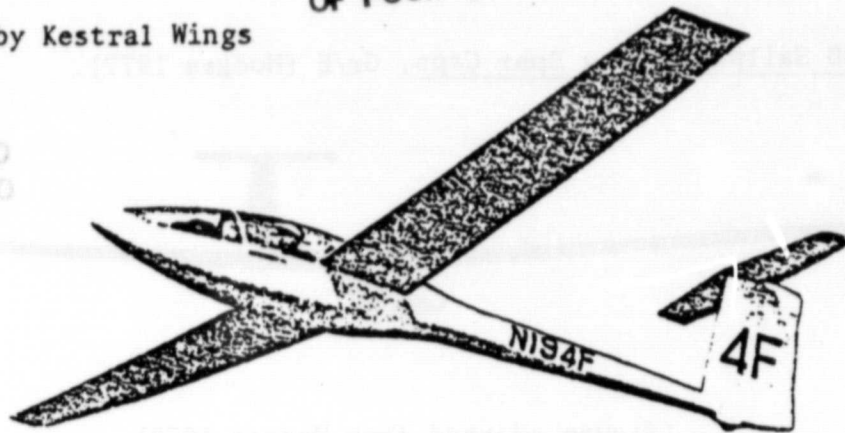


(figure adapted from Hodges 1977)

Slingsby Kestrel Sailplane Wings, primary, British competitive sailplane, Gr/E spar caps, 50% lighter than Gl/E they replaced, Gl skins, fatigue tested sub-components, the only beam to fail was cycled +7.5 to -6g's for 110,000 cycles, first used in 1971, has accumulated over 1,000 hours flight service, reportedly first use of Gr fibers in primary structure in general aviation (Hodges 1977).

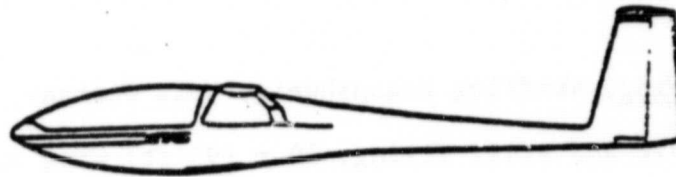
Slingsby Kestral Wings

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(figure adapted from Hodges 1977)

Slingsby Vega Sailplane Wing Spar Caps, Gr/E (Hodges 1977).



(figure adapted from Hodges 1977)

Spacecraft Miscellaneous Composite structure has been developed and applied to many spacecraft. The following list is by no means complete but is included to give a flavor of spacecraft uses:

1. Space Shuttle

- (a) leading edge is C/C converted to silicon carbide for oxidation resistance at free stream gas temperatures of 3500°F
- (b) truss for engine thrust structure is B/E, and is 30% lighter than titanium
- (c) aft body flap, Gr/Pi, 6.4×2 m (21×7 ft), 162 kg (356 lb)

2. Applications Technology Satellite Truss Tubes, 4.6 m (15 ft) tubes, 36 kg, Gr/E, 50% weight savings.

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3. Viking Orbiter Antennae, Gr/E
4. ANIK communications satellite
5. Subreflector on an antenna

More applications are reported in the references (Salkind 1976, Lenoe and Neal 1975, Wesselski 1976, Berman 1975, Maxwell, et al. 1975, Advanced Composites Design Guide, Third Edition 1973 Vol. V, Section 5.3, Miller and Robertson 1977).

Stander BS-1 Sailplane Miscellaneous Parts (Hodges 1977).



(figure adapted from Hodges 1977)

S-3 (Viking) Spoiler, Navy, Gr/E skins on honeycomb, about  $.3 \times 1.8$  m ( $1 \times 6$  ft), 20 parts are on 10 aircraft for 5 years service evaluation, approximately 4 years into program now, no problems have been found, ground handling has damaged parts, repairs have been successfully made (personal communication with Dr. M. S. Rosenfeld).

T2A Horizontal Stabilizer, reportedly a demonstration that reinforced composites can provide in some cases a degree of inherent fail-safety, 85% of damage causing load was sustained after damage occurred, whereas metal structure sustained only 20% of its damage causing load after damage occurred (Hackman 1973).



(figure adapted from  
Hackman 1973)

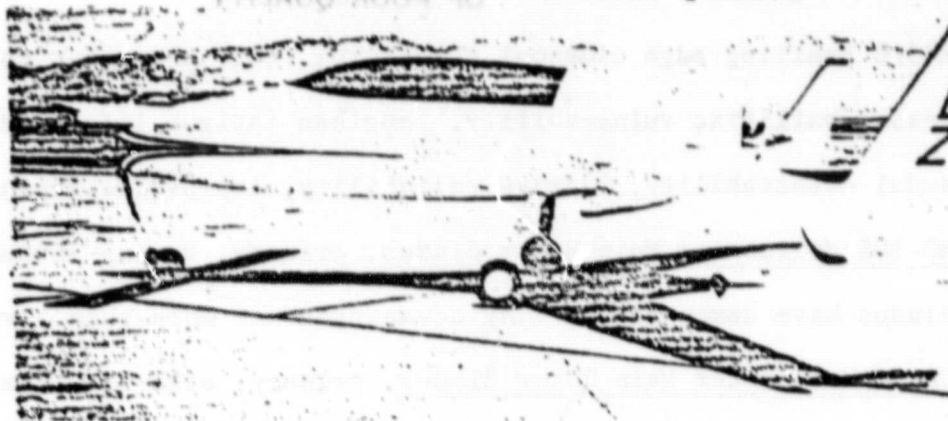
T-2B Wing, S-GI filament wound, development program included developing design allowables for unidirectional and fabric S-GI, specimens and structural elements tested, full scale wing was 40% lighter, met strength and stiffness requirements, static test failure was 165% limit load, costs were shown to be competitive with metal (Hackman 1973).

T-39 Center Section Wing Box, Rockwell International, USAF, two boxes were built and tested, one statically at RI and the other in fatigue at WPAFB, potentially 37% lighter, static bending failure at 60% ultimate load, failure originated by stress concentration in corner of outer face sheet of compressive cover at intersection of splice plate and panel edge, plate provided lateral restraint to intersecting composite material due to large difference in Poisson's ratio, fatigue test developed from 150,000 hour, 3.0 life multiplying factor, no apparent damage to part (Hackman 1973).

Windmill Blade, "Command 150 ft", NASA, Kaman Industries, leading edge E-GI/PE filament wound D spar, trailing edge two panels, each panel a sandwich of GI/Resin fabric skins adhesively bonded to phenolic resin impregnated paper honeycomb core, failure occurred during static test at 106% design limit load whereas it was expected to fail at above 150% limit load, failure has been possibly attributable to fabrication irregularities (bumps and reduced wall thickness) which led to compressive buckling of the D spar (personal communications with NASA LeRC personnel).

Vari-Eze, primary, Burt Rutan 2 place homebuilt first flew in 1975, wing and fuselage were GI/E skins built up on urethane foam core which has been cut to shape, eliminating need for large molds (Hodges 1977).

Vari-Eze

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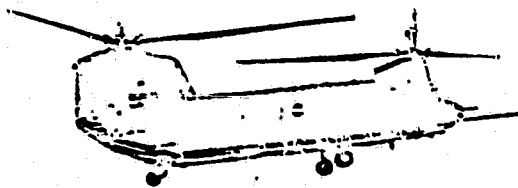
(figure adapted from Hodges 1977)

VTOL (Vertical Take-Off and Landing) Aircraft Miscellaneous. Many primary and secondary parts including main and tail rotor blades, drive shafts, fuselage parts, and fairings have been developed and in many cases put into production. Rotor blades on helicopters have generally outperformed metal blades due to overall cost savings, low notch sensitivity, high impact strength, gradual and non-catastrophic damage accumulation, corrosion resistance, high fatigue strength, ease of manufacturing compound curved geometries, inherent structural damping, and repairability. Composite rotor blades have proven to be particularly tolerant of ballistic damage. Due to improved notch sensitivity, ballistic impact which would cause catastrophic failure in metal blades often results only in loss of some composite blade cross section and not in failure. The following applications of composites to VTOL aircraft have been made; more are reported in the references. (Salkind 1973, Hackman 1973, Salkind 1976, Homies 197x, Advanced Composites Design Guide, Third Edition 1973 Vol. V, Section 5.2, personal communication with Boeing Vertol, Hofstedt 1976, Grina 1975).

1. AH-1 (Cobra) Helicopter Main Rotor Blades, primary, Bell Helicopter Company/KAMAN Aerospace Company/Hercules Incorporated/Structural Composites Industries/Army, hollow D-section spar of unidirectional E-Gl/E, fabric Gl/E afterbody of rib and skin construction, first flight in January 1961, later developments include use of filament wound S-Gl/E spars, Nomex core trailing edge,

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- K49/E trailing edge closeout structure, tests show that composite blades lessen ballistic vulnerability, lengthen fatigue life, reduce radar and audal detectability, improve reliability, improve maintainability (White 197x).
2. BO-105 Helicopter Main Rotor Blades, primary, Boeing Vertol Company, G1/E blades have demonstrated many advantages for more than 7 years (Holmes 197x).
  3. CH-46 Helicopter Main Rotor Blades, primary, each is 7.8 m (25.5 ft), 70.5 kg (155 lbs) (Gardner and Thompson 1978).
  - 4a. CH-47 Helicopter Main Rotor Blades, primary, Boeing Vertol Company, early on a B/E blade set was developed and flown 2,000 hr, however G1/E is primarily used in production with some Gr/E for cost effectiveness, 1963 (Salkind 1976).



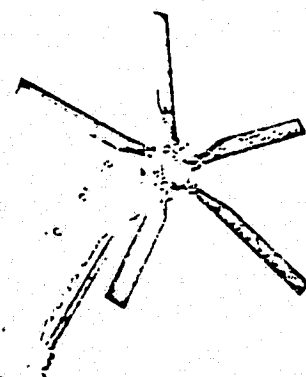
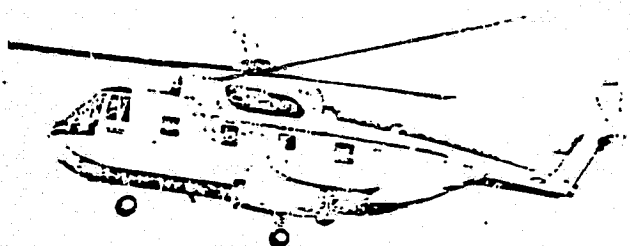
(figure adapted from Pickney 1973)

- 4b. CH-47 Helicopter Miscellaneous Parts, secondary, Boeing Vertol Company, some miscellaneous parts such as the upper fairings on the fuselage around the rotor shaft are G1/E skins on Al honeycomb cores, some use 181 style E-G1/E.
- 5a. CH-54B (Flying Crane) Helicopter Aft Fuselage Stringer Reinforcements, Sikorsky/Army, unidirectional B/E reinforcement strips are bonded to the vertical legs of 12 Al airframe stringers running along the top and bottom skins of the tail cone (aft fuselage), the initial metal design had a resonant bending problem with certain combinations of payload weight and sling length, stringer reinforcement 5 m (16.4 ft) long, 180 kg (397 lb)/aircraft, 14% weight savings, one has been in Army service since 1972 and receives periodic



inspections, service has been satisfactory, this aircraft accumulates flight time rather slowly, development tests included static strength, stiffness, and fatigue (Salkind 1976, Pride 1978, Hardrath 1977).

- 5b. CH-54B (Flying Crane) Helicopter Tail Skid Tubes, secondary, Sikorsky/Army, B/E tubes bonded and bolted to metal end fittings form a truss for the tail skid, and have also been in service since 1972 (Salkind 1976).
- 6. HH-43 Helicopter Main Rotor Blades, primary.
- 7a. HLH (Heavy Lift Helicopter) Main Rotor Blades, primary, Boeing Vertol Company/Army, blades are 12.8 m (42 ft) long, 343 kg (755 lbs), 8 blades per set.
- 7b. HLH (Heavy Lift Helicopter) Miscellaneous Parts, Boeing Vertol Company/Army, fuel pods are being made using K/E and G1/E skins on Nomex honeycomb core.
- 8. H-3 Helicopter Main Rotor Blades, primary, Sikorsky/Navy, blades of  $0^\circ$  G1/E and  $\pm 45^\circ$  Gr/E are made in two longitudinal halves and then joined.
- 9. H-13 Helicopter Main Rotor Blades, primary, Bell .
- 10. H-21 Helicopter Main Rotor Blades, primary, Piasecki, in mid 1950's Boeing/USAF also studied composite fuselage parts.
- 11. H-23 Helicopter Main Rotor Blades, primary, Hiller.
- 12. H-53 Helicopter Cockpit Canopy, secondary, cockpit framework around windows is a large complex curved structure and G1/E is 50% less costly to manufacture, less production joining manhours are required.
- 13. OH-6A Helicopter Main Rotor Blades, primary, Hughes/Army, experimental blades.
- 14. SA-341 Rotor Blades, primary.
- 15. S-61 Helicopter Tail Rotor Blade and Drive, Sikorsky.



(figure adapted from Salkind 1973)

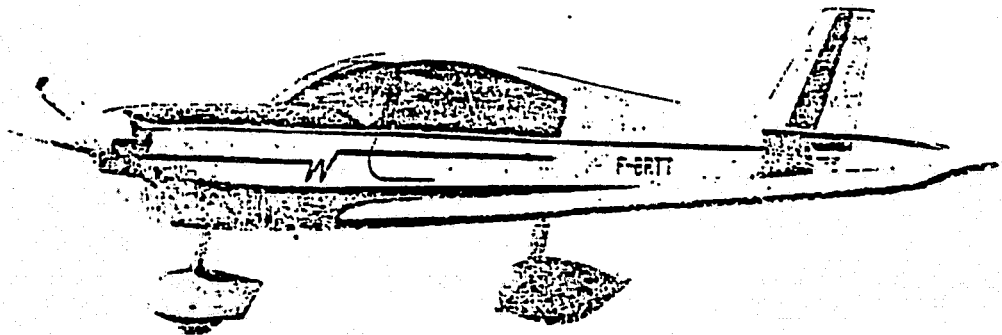
16. UH-1 Helicopter Tail Rotor Blades, Kaman, a bearingless tail rotor blade.



(figure adapted from Salkind 1973)

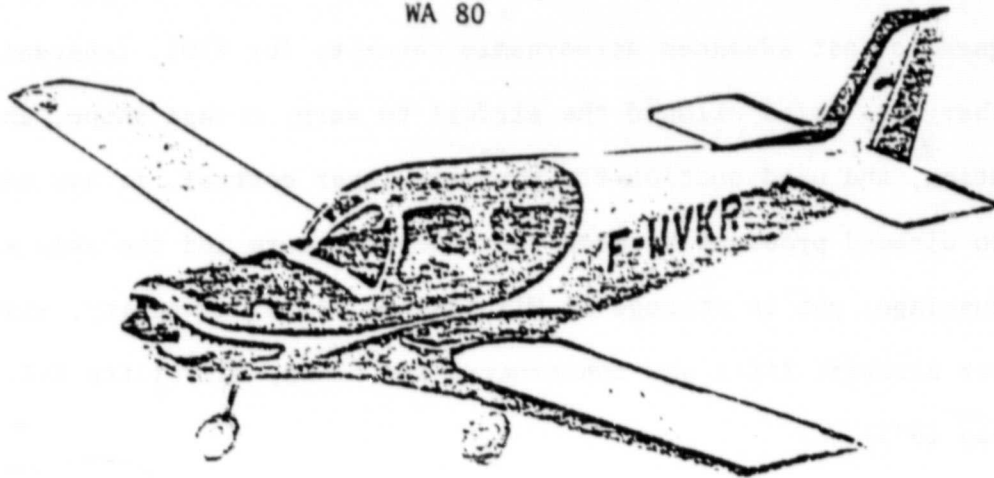
17. VFW-40U VTOL Drive System Shafts, primary.
18. VS-300 Helicopter Main Rotor Blades, Sikorsky.
- 19a. YUH-60A UTTAS Helicopter Tail Rotor Blades, Sikorsky/Army, the bearingless tail rotor has a continuous unidirectional Gr/E spar extending from tip of one blade, through hub, to tip of opposite blade, spar has high centrifugal strength and bending stiffness, but low torsional stiffness so that a torque tube can change blade pitch by twisting the spar, without need for a bearing, hub is lighter due to reduced centrifugal load, rotor system is 40% lighter than conventional, hinged rotor system (Salkind 1976).
- 19b. YUH-60A UTTAS Helicopter Fairings, Sikorsky/Army, more than 50 K/E fairings used.

WA-51 (Pacific), primary, Wassmer in France, began building this lightplane in 1970, reportedly all composite except for some carry-through structure at the wing-fuselage interface, also WA-80 Pirahna in production, delivered about 30 lightplanes/year in early 1970's (Hodges 1977).



(figure adapted from Hodges 1977)

WA 80



(figure adapted from Hodges 1977)

Windecker Eagle, primary, Windecker, FAA certified in 1979 (Hodges 1977).

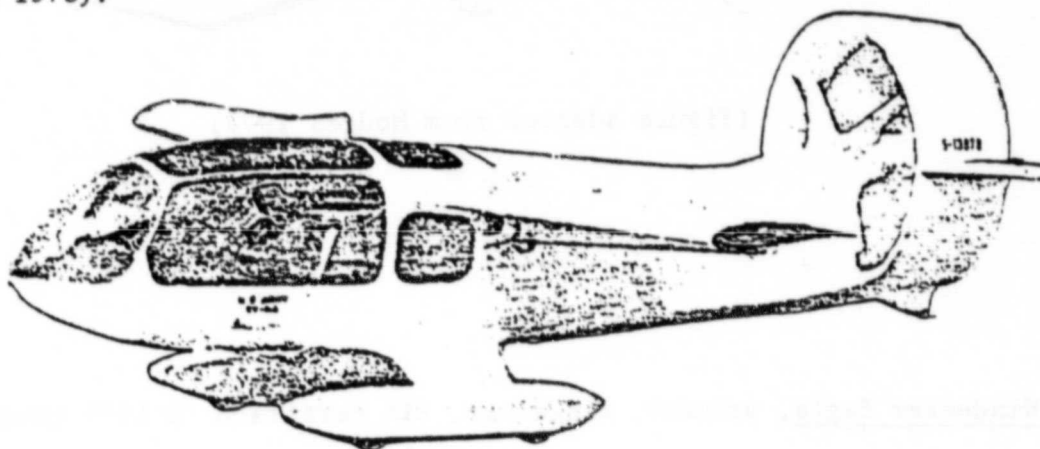


(figure adapted from Salkind 1976)

XFV-12A Prototype Wing Box, Navy, V/STOL prototype aircraft, a Gr/E with nomex honeycomb core wing to fuselage box survived 3 lifetimes on ground test, next upper skin FOD excitation will be introduced to and tested for damage to fatigue strength (personal communications with NADC personnel).

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XV-11, (Marvel P.), primary, Mississippi State University, U.S. Army, assembled in 1965, most of the structure was G1/PE with steel and aluminum reinforcements, designed to test advanced aerodynamic concepts for STOL, inherent flexibility of fiberglass skins allowed the airfoil to warp, change shape, and camber distribution, and used suction for boundary layer control, it has ceased flying due to disbond problems between the honeycomb core and the skin at the rear of the fuselage, put in storage at Mississippi State University, also another earlier aircraft XAZ-1 was constructed and flown, Marvelitte XAZ-1 (Hodges 1977, Hackman 1973).



(figure adapted from Hodges 1977)

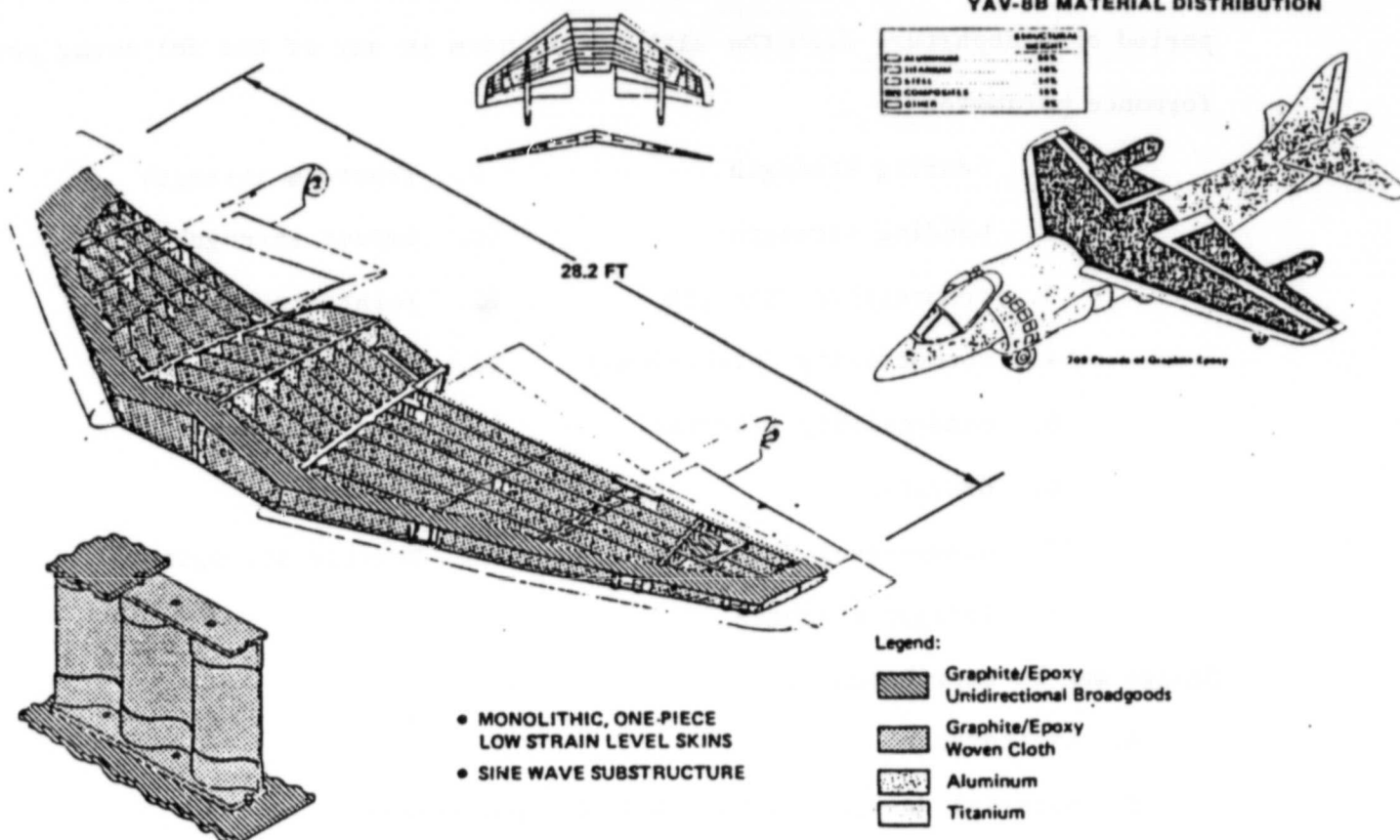
YAV-8B (Harrier) Wing, primary, 419 kg (921 lb) total weight, 8.6 m (28.2 ft) span, consists of

200 kg	438 lb	47%	Gr/E unidirectional
86 kg	190 lb	21%	Gr/E woven
57 kg	125 lb	14%	Aluminum
44 kg	96 lb	10%	Titanium

monolithic, one piece skins with sine wave substructure, NADC test 5 different representative proposed wing box sections, and found that the fatigue, impact, and moisture excitations requires a strain limit which imposes a weight penalty, resulting in 19% weight savings, scheduled to have flown by publication of this report (Siegel 1977, Weinberger, et al. 1977, Riley 1978, Huttrop 1978).

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### AV-8B HARRIER COMPOSITE WING



(figures adapted from Siegal 1977)

YC-14 (Advanced Medium STOL Transport) Horizontal Stabilizer, Boeing/USAF development prototype, test parts have been constructed (personal communication with Boeing).

YF-17 Miscellaneous Parts, Northrop, 409 kg (900 lb) of Gr/E and 91 kg (200 lb) of K/E used on this prototype in primary and secondary structure, including wing skins, empennage, strakes, and access panels, most damage has been attributed to ground maintenance, B/E landing gear link weighing 25% less, problems including extruding fasteners through composite panels, dropped tools causing localized damage (Salkind 1976, personal communication with USAF personnel).

## APPENDIX B

### VOCABULARY DEFINITIONS

Damage is the occurrence of a measureable change over an arbitrary time period or a departure from the anticipated norm in any of the following performance parameters\*:

- |                              |                       |
|------------------------------|-----------------------|
| 1. bearing strength          | 9. fracture strength  |
| 2. bending strength          | 10. impact strength   |
| 3. compressive strength      | 11. joint strength    |
| 4. conductivity (electrical) | 12. notch sensitivity |
| 5. conductivity (thermal)    | 13. shear strength    |
| 6. density                   | 14. stiffness         |
| 7. dimensional stability     | 15. tensile strength  |
| 8. fatigue strength          |                       |

Damage may be attributed to:

- A. Excitations
- B. Material design and manufacturing parameters
- C. Flaws
- D. Combinations of A, B, and C.

Environment is the combination of all excitations to which a structure\* is potentially sensitive, and to which damage may be attributed. In the literature, a somewhat narrower useage of the term is typically found. It often is used to denote only hygrothermal excitations. This narrow useage of the term unfortunately often leads to (unintentional) exclusion of other excitations from the reporting procedure.

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\*In this context, a test coupon or specimen is viewed as a structure as much as an entire fuselage assembly.

Excitation is an event or combination of simultaneous or sequential events to which a part or specimen is subjected after manufacture. Excitation may or may not cause damage.

1. Mechanical Loading

A. Monotonic

1. Tension
2. Compression
3. Bending
4. Shear

B. Cyclic

1. TT
2. CC
3. TC
4. Bending
5. Shear

C. Impact

D. Service Conditions

2. Chemical (non-H<sub>2</sub>O)

3. Moisture\* (H<sub>2</sub>O)

4. Corrosion

5. Erosion

6. Radiation

7. Temperature\*

A. Effect on material property

B. Effect of thermal gradient

8. Time (aging, frequency, load rate)

Failure is the inability of a structure to perform its intended function (excluding cosmetic rejections) as a result of damage.

Flaws are undesired but often discernable irregularities and/or discontinuities in the laminate or structure. Flaws may be built in during manufacture, and/or may accumulate due to initiation and/or growth from service excitations. Flaws themselves are not damage, however, flaw accrual may or may not cause damage. Flaws (Hoffman and Konishi 1977 contains a complete listing) include:

\*Note that combinations of moisture and thermal excitations are also termed hygrothermal.

cracks	improper interface properties
crazing	improper fiber properties
delaminations	resin rich areas
voids	resin starved areas
missing lamina	improper fiber/matrix ratio
misoriented lamina	breakaway plies around holes
improper stacking sequence	mislocated holes
improper matrix properties	missized holes
improper thickness, shape or other dimensions	undersized fasteners
inclusions (hair, dirt, etc.)	NDE (Q/A) indications of uncertain nature
improper surface finish	wrinkles
debonds from honeycomb	lumps
debonds from any joint surface	

Laminate Analysis is the process of predicting laminate response to given excitations.

Macromechanical Models assume the lamina is homogeneous and attempt to model material behavior on the lamina or laminate level. Most often, each lamina is assumed to be anisotropic and in particular orthotropic. These mathematical models may be analytically, semiempirically, or empirically derived. A more detailed discussion given by Jones, 1975, pp. 31-84 and pp. 147-238, although we use a slightly different definition.

Materials (Laminate) Design is the process of designing the laminate, which requires selecting fiber and matrix types, stacking sequence, orientation angles, and processing which will efficiently carry the specific magnitudes and directions of structural loads and other excitations.

Material Design and Manufacturing Parameters are the composite constituent assembly and processing parameters which the designer and manufacturer can quantify or infer and choose and/or control within some ranges. Examples include:

Fiber Related

- length
- diameter
- surface finish
- material properties
  - modulii
  - yield strength
  - ultimate strength

\*Performance Parameters 1-15 are arbitrarily and loosely defined in a manner which facilitates categorizing routinely encountered engineering literature. As a matter of convenience, they are listed alphabetically, not by order of importance.



conductivity  
adsorptivity  
fatigue strength  
efficient of thermal expansion  
woven  
non-woven

Matrix Related

modulus  
yield strength  
ultimate strength  
endurance strength  
deflection temperature  
glass trans. temperature  
conductivity  
adsorptivity  
coefficient of thermal expansion  
shear strength  
fracture toughness

# plies  
orientation angles  
stacking sequence  
hybrid  
volume ratio  
heat treatment  
laminating pressure  
interface strength  
cure cycle

Geometric (machined)

holes  
notch  
cutout  
edge

Micromechanical Models assume lamina heterogeneity and anisotropy and attempt to predict lamina properties based upon interaction of constituent fiber and matrix properties. These mathematical models may be either analytically or semiempirically derived. A more detailed discussion given by Jones, 1975, pp. 85-147, although we use a slightly different definition.

Primary Structure is any part of assemblage of parts of flight hardware whose failure would typically result in total inability to continue safe and controlled flight and/or landing.

Secondary Structure is any part of assemblage of pieces of flight hardware whose failure would be expected to handicap continued safe and controlled flight and/or landing, to limited or minor extent.

Structural Analysis is the process of predicting response of a given structural configuration under given excitations, usually mechanical.

Structural Design is the process of selecting structural configurations, materials (laminates), and dimensions to satisfy a set of structural design factors and their associated structural design criteria. The criteria may be changed as tradeoffs may arise. For example, cost criteria may be relaxed somewhat to allow for improvements in joint strength.

Structural Design Criteria are quantified constraints placed upon the structural design factors which a structure must satisfy. Often, tradeoffs in criteria may be necessary.

Structural Design Factors are the characteristics or considerations which influence the design of the structure. Usually a number of structural design factors must be considered in a given design situation. However, one or two are often critical; and when they are satisfied, the remaining factors may be automatically satisfied. Shigley 1972 reports the following factors:

- |                           |                    |
|---------------------------|--------------------|
| 1. Strength               | 12. Noise          |
| 2. Reliability            | 13. Styling        |
| 3. Thermal Considerations | 14. Shape          |
| 4. Corrosion              | 15. Size           |
| 5. Wear                   | 16. Flexibility    |
| 6. Friction               | 17. Control        |
| 7. Processing             | 18. Stiffness      |
| 8. Utility                | 19. Surface finish |
| 9. Cost                   | 20. Lubrication    |
| 10. Safety                | 21. Maintenance    |
| 11. Weight                | 22. Volume         |

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# APPENDIX

## WU'S DATA REDUCTION

That  $K_{Ic}$  is a material constant was demonstrated (Wu 1968) by plotting  $\log \sigma_c^\infty$  vs.  $\log a_c$ . If one considers for example the equation

$$K_{Ic} = \sigma_c^\infty \sqrt{a_c} \quad (D1)$$

where  $K_{Ic}$  = critical stress intensity

$\sigma_c^\infty$  = farfield stress

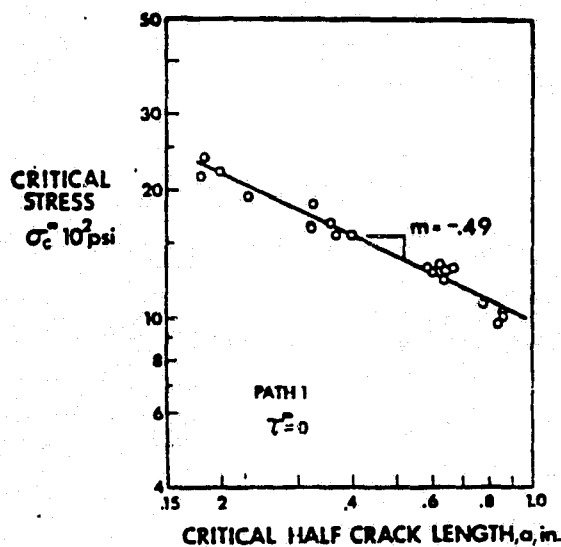
$a_c$  = critical flaw dimension

which may also be expressed

$$\log K_{Ic} = \log \sigma_c^\infty + \frac{1}{2} \log a_c \quad (D2)$$

then if  $K_{Ic}$  is a material constant, the plot of  $\log \sigma_c^\infty$  vs  $\log a_c$  will have a slope of  $-\frac{1}{2}$  if it obeys eq. D1. Similar results are available for  $K_{2c}$  (Wu 1968).

### Cracked Plate Under Tension



(Adapted from Jones 1975 and Wu 1968)

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